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EVALUATION OF UNDEVELOPED ROCKET ENGINE CYCLE APPLICATIONS TO ADVANCED TRANSPORTATION

FINAL REPORT

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SRS TECHNOLOGIES

SYSTEMS TECHNOLOGY DIVISION
990 EXPLORER BLVD. N.W.
CUMMINGS RESEARCH PARK WEST
HUNTSVILLE, ALABAMA 35806
(205) 895-7000

FOREWORD

This 12-month effort was performed by SRS Technologies, with Dr. Richard D. Kramer being the SRS Project Manager. The Acurex Corporation, a subcontractor to SRS, provided data and analyses on the full flow staged combustion cycles. Mr. Bruce Wiegmann of the NASA-Marshall Space Flight Center was the COTR, and was assisted by Mr. Robert Champion, who served as the Technical Advisor. The following personnel contributed to this effort:

Mr. James Pearson, Jr. - SRS
Mr. Luke Spears - SRS
Mr. Mike McLeod - SRS
Mr. Dick Steinberg - SRS
Mr. William Knuth - Acurex (Project Manager)
Mr. John Beverage - Acurex.

Richard D. Kramer

Richard D. Kramer, Ph.D.
Project Manager

SUMMARY

Undeveloped pump-fed, liquid propellant rocket engine cycles were assessed and evaluated for application to Next Manned Transportation System (NMTS) vehicles, which would include the evolving Space Transportation System (STS Evolution), the Personnel Launch System (PLS), and the Advanced Manned Launch System (AMLS). Undeveloped engine cycles selected for further analysis had potential for increased reliability, enhanced maintainability, reduced cost, and improved (or possibly level) performance when compared to the existing SSME and proposed STME engines.

The split expander (SX) cycle, the full flow staged-combustion (FFSC) cycle, and a hybrid version of the FFSC, which has a LOX expander drive for the LOX pump, were selected for definition and analysis. Technology requirements and issues were identified and analyses of vehicle systems weight deltas using the SX and FFSC cycles in AMLS vehicles were performed. A strawman schedule and cost estimate for FFSC subsystem technology developments and integrated engine system demonstration was also provided.

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1.0 INTRODUCTION

The next generation of manned space transportation systems are being studied and analyzed with an increased emphasis on reducing the total Life-Cycle-Cost (LCC) of manned access and payload delivery to earth orbit. In the formulation of the Next Manned Transportation System (NMTS), propulsion is a key area for cost reduction because of the high cost elements associated with the current Space Transportation System (STS). The objectives of this task were to address new and innovative chemical propulsion concepts which are not currently being considered as candidate propulsion options for NMTS vehicles, to compare them with the current baseline propulsion concepts for NMTS, and to identify and define the technology programs necessary to implement these new and innovative concepts.

Efforts on this task were concentrated on the evaluation of liquid hydrogen, liquid oxygen, pump-fed engine systems that showed promise for improvements in vehicle performance and significant reductions in costs. System design, development, manufacture, and operational features were qualitatively addressed. In addition, engine system reliability, maintainability, and low-cost potential were evaluated to determine the most promising concepts for further consideration. The task was organized into subtasks as shown in Figure 1.0-1.

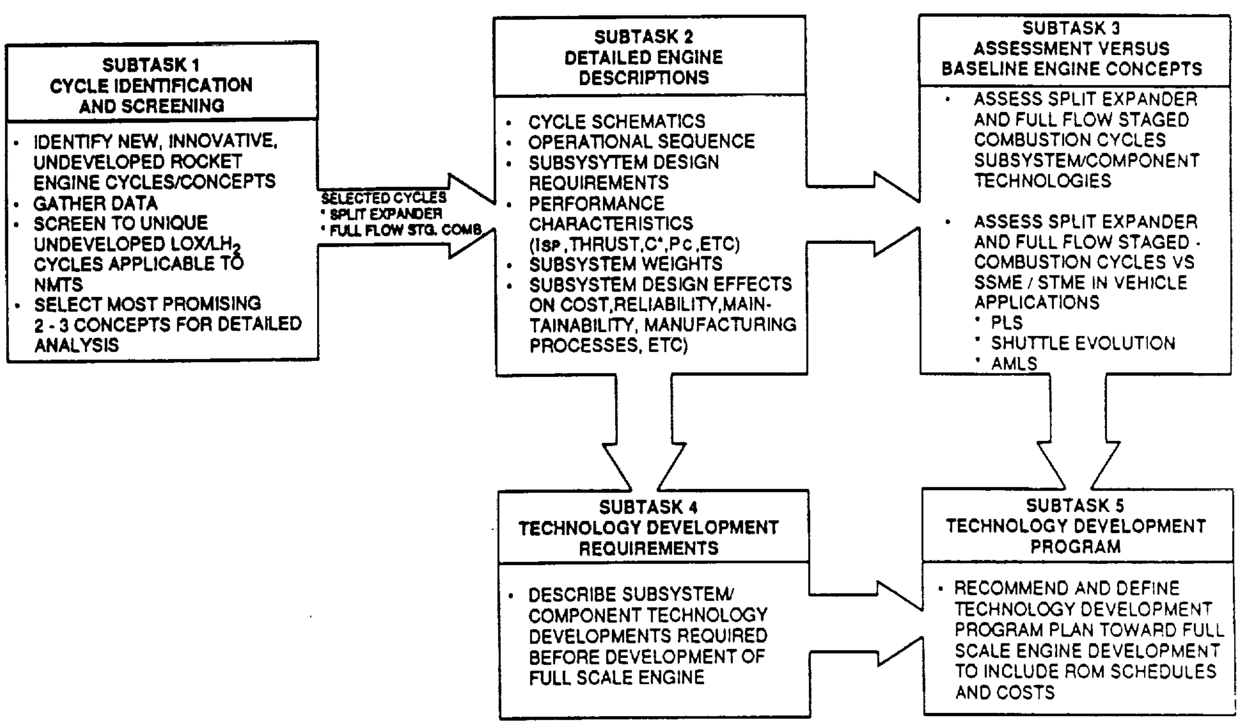


FIGURE 1.0-1 TASK FLOW

2.0 ANALYSIS AND RESULTS

The analyses performed and their results are described in the following sections.

2.1 Engine Cycle Identification and Screening (Subtask 1)

Undeveloped engine cycles were identified and evaluated to determine the most promising candidates for further definition. A literature search was performed to gather the available information on pump-fed undeveloped rocket engine cycles and advanced concepts. A large amount of data was gathered on advanced/undeveloped cycles (gas generator, staged combustion, expander, full flow, etc.). Information contained in past engine studies involving advanced oxygen/hydrogen and oxygen/hydrocarbon engine concepts, including aerospike/plug nozzles, dual-fuel/tripropellant, and dual throat/dual expander concepts, was also compiled.

Because of the large number of different cycle/concept variations, an undeveloped cycles data base was developed using R-BASE on an IBM-AT. This allowed the data to be exported to files in other formats such as SYMPHONY. A total of 114 entries were included in the data base as shown in Figure 2.1-1. Since the objective of this task was not to generate a complete data base, but to assess undeveloped cycles, only general descriptive performance parameters were gathered in the initial search.

After gathering basic performance data on the 114 undeveloped cycles/advanced concepts, a prescreening process was initiated to enable an in-depth look at the most promising concepts/cycles. The prescreening qualitative criteria for the undeveloped cycles data base is shown in Figure 2.1-2. With NASA/MSFC concurrence, a total of nine out of the 114 cycles/concepts were chosen for further assessment, as shown in Table 2.1-1.

2.1.1 Generic Cycle Assessment

An independent assessment of generic rocket engine cycles was also made. The generic cycles assessed were divided into three major groups, gas generator cycles, expander cycles, and staged combustion cycles. Options under each generic cycle group were assessed as shown in Table 2.1-2.

Entry #	Cycle/Propellant/Source
1	Staged Combustion/Hybrid PROPELLANT: LOX/Hydrogen LPIAG 1/26/89
2	Dual Fuel/Dual Bell, Boost phase engine PROPELLANT: CH4/O2 STME Configuration Study, Pratt & Whitney, July 1987
3	Dual Fuel/Dual Bell, Boost phase engine PROPELLANT: H2/O2 STME Configuration Study, Pratt & Whitney, July 1987
4	Dual Fuel/Dual Bell, Orbit phase engine PROPELLANT: H2/O2 STME Configuration Study, Pratt & Whitney, July 1987
5	Dual Fuel, Single Bell (Option 2), Boost Phase Engine PROPELLANT: H2/O2 STME Configuration Study, Pratt & Whitney, July 1987
6	Dual Fuel, Single Bell - (Option 2) Orbit Phase Engine PROPELLANT: H2/O2 STME Configuration Study, Pratt & Whitney, July 1987
7	High/Variable Mixture Ratio PROPELLANT: O2/H2 STME Configuration Study, Pratt & Whitney, July 1987

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS

Entry #	Cycle/Propellant/Source
8	STME Split Expander Engine
	PROPELLANT: H2/LO2
	STME & STBE Configuration Study, P & W, 3/29/89
9	Derrivative STBE - Split Expander Cycle
	PROPELLANT: H2/LO2
	STME & STBE Configuration Study, P & W, 3/29/89
10	Derrivative STBE - Split Expander Cycle
	PROPELLANT: CH4/LO2
	STME & STBE Configuration Study, P & W, 3/29/89
11	Split Expander LO2/LH2 for LRB
	PROPELLANT: LO2/LH2
	LRB Systems Study, 2/3/89 General Dynamics
12	LO2/LH2 Split Expander
	PROPELLANT: LO2/LH2
	LRB Systems Study, 10/6/88, General Dynamics
13	Split Expander
	PROPELLANT: LO2/LH2
	Aerojet STME/STBE Configuration Study, 3/22/89
14	STME - Split Expander Alternate - Core Stage
	PROPELLANT: LO2/LH2
	STME & STBE Quarterly Program Review,3/21/89 Rocketdyne
15	STME Split Expander - 2 Pc nozzle alt - booster stage
	PROPELLANT: LO2/LH2
	STME/STBE Quarterly Program Review, 3/21/89 Rocketdyne

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
16	High/Variable Mixture Ratio Oxygen/Hydrogen Engine PROPELLANT: LOX/LH2 all Adv ETO Propulsion Technology, 1988, Vol 1 see other
17	High Variable Mixture Ratio O2/H2 Engine PROPELLANT: O2/H2 Adv ETO Propulsion Technology, 5/12/88,Vol 1, see other
18	High/Variable Mixture Ratio O2/H2 PROPELLANT: -0- R. Beichel, C.J. O'Brien, and E.K. Bair, see other
19	Expander Cycle for Plug Cluster PROPELLANT: LO2/LH2 Unconventional Nozzle Tradeoff Study, Aerojet Liquid Rocket Company, July 1979, NAS 3-20109 for NASA LeRC
20	Gas Generator Cycle Plug Cluster Model I PROPELLANT: LO2/LH2 Unconventional Nozzle Tradeoff Study, Aerojet Liquid Rocket Company, July 1979 NAS 3-20109 for NASA LeRC
21	Engine cycle concepts and relative evaluation-see other PROPELLANT: -0- Rocketdyne, Advanced Propulsion for Space Sortie Vehicle, Technical proposal, 7/19/83
22	Dual-Fuel/Dual-Throat,baseline staged combustion exampl PROPELLANT: LO2/RP-1, LO2/LH2, LO2/RP Dual-Fuel,Dual-Throat Engine Preliminary Analysis,Final Report,Aerojet Liq. Rocket Company,Aug 1979 NAS8-32967
23	Split Combustor Linear Aerospike Engine - Concept 1 PROPELLANT: Outer combustor - O2/RP-1 Linear Aerospike Engine Study, Final Report, Rocketdyne, NAS3-20114, November 1977

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
24	Split Combustor Linear Aerospike Engine - Concept 2 PROPELLANT: Outer combustor - O2/H2,Inner combustor- O2 Linear Aerospike Engine Study, Final Report, Rocketdyne, NAS3-20114, November 1977
25	Staged Combustion Baseline Mode 1 PROPELLANT: Oxygen,RP-1 Advanced Propulsion System Concepts for STS, Aerojet Liquid Rocket Company, October 18 & 19 1976
26	Gas Generator Alternate Mode 1 PROPELLANT: O2/RP-1 Advanced Propulsion System Concepts for STS, Aerojet Liquid Rocket Company, October 18 & 19 1976
27	Dual Fuel Staged Combustion Mode 1 PROPELLANT: Oxygen, RP-1 Advanced Propulsion Concepts for STS, Aerojet Liquid Rocket Company, October 18 & 19, 1976
28	Dual Fuel Staged Combustion Mode 2 PROPELLANT: Oxygen/LH2 Advanced Propulsion Concepts for STS, Aerojet Liquid Rocket Company, October 18 & 19, 1976
29	Tri-Propellant Engine, Series Burn, Mode I & Mode II PROPELLANT: LO2/RP-1, LO2/LH2 Advanced Propulsion System Concepts for STS, Aerojet Liquid Rocket Company, October 18 & 19, 1976
30	Tri-Propellant, Parallel Burn, Mode I & Mode II PROPELLANT: LO2/RP-1+LH2, LO2/LH2 Advanced Propulsion System Concepts for STS, Aerojet Liquid rocket Company, October 18 & 19 , 1976
31	several studied - see other PROPELLANT: -0- Engine Parametrics for Mixed-Mode Single Stage to Orbit Vehicles, Aerojet Liquid Rocket Company, NAS3-19727

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
32	-0- PROPELLANT: -0- Advanced Oxygen-Hydrogen Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
33	LO2/RP-1 fuel rich (FR) gas generator PROPELLANT: LO2/RP-1 with RP-1 coolan Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
34	LO2/RP-1 fuel rich (FR) gas generator PROPELLANT: LO2/RP-1 with LO2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
35	LO2/CH4 FR GG cycle PROPELLANT: LO2/LCH4 with LCH4 coolan Advanced Oxygen-Hydrogen Rocket Engine Study, April 1981, NAS 8-33452
36	LO2/RP-1 FR preburner (PB) staged combustion (SC) cycle PROPELLANT: LO2/RP-1 with RP-1 coolan Advanced Oxygen-Hydrogen Rocket Engine Study, April 1981, NAS 8-33452
37	LO2/RP-1 FR PB SC cycle PROPELLANT: LO2/RP-1 with LO2 coolant Advanced Oxygen-Hydrogen Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
38	LO2/RP-1 oxidizer rich (OR) PB SC cycle PROPELLANT: LO2/RP-1 with RP-1 coolan Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid rocket Company, April 1981, NAS 8-33452
39	LO2/RP-1 OR PB SC cycle PROPELLANT: LO2/RP-1 with LO2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid rocket Company, April 1981, NAS 8-33452

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
40	LO2/LCH4 FR PB SC cycle PROPELLANT: LO2/LCH4 with LO2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid rocket Company, April 1981, NAS 8-33452
41	LO2/LCH4 FR & OR PB SC cycle PROPELLANT: LO2/LCH4 Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
42	LO2/LH4 FR GG cycle PROPELLANT: LO2/RP-1 with LH2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
43	LO2/LH4 FR GG & LO2/LCH4 OR PB mixed cycle (GG & SC) PROPELLANT: LO2/LCH4 Dual Throat with Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
44	LO2/LH4 FR GG & LO2/RP-1 OR PB mixed cycle (GG & SC) PROPELLANT: LO2/RP-1 Dual Throat with Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
45	LO2/LH2 FR PB SC cycle PROPELLANT: LO2/RP-1 with LH2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
46	LO2/LH2 FR PB & LO2/RP-1 OR PB SC cycle PROPELLANT: LO2/RP-1 with LH2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
47	LO2/LH2 FR PB & LO2/RP-1 FR & OR PB SC cycle PROPELLANT: LO2/RP-1 with LH2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
48	LO2/LH2 OR GG cycle PROPELLANT: LO2/RP-1 with LH2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
49	heated H2 expander bleed (EB) cycle PROPELLANT: LO2/RP-1 with LH2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
50	heated H2 & LO2/Rp-1 OR PB mixed cycle (EB & SC) PROPELLANT: LO2/RP-1 with LH2 coolant Advanced Oxygen-Hydrocarbon Rocket Engine Study, Aerojet Liquid Rocket Company, April 1981, NAS 8-33452
51	Staged Combustion - Oxygen cooled PROPELLANT: RJ-5/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141
52	Staged Combustion - RJ-5 cooled PROPELLANT: RJ-5/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141
53	Staged Combustion (Parallel Burn) - H2 cooled, Mode I PROPELLANT: RJ-5/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141
54	Gas Generator Cycle - H2 cooled, Mode I PROPELLANT: RJ-5/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141
55	Staged Combustion Cycle - RP-1 Cooled, Mode I PROPELLANT: RP-1/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
56	Staged Combustion Cycle - Hydrazine Cooled, Mode I PROPELLANT: Hydrazine/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141
57	Staged Combustion Cycle - MMH Cooled, Mode I PROPELLANT: MMH/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141
58	Staged Combustion Cycle - CH4 Cooled, Mode I PROPELLANT: CH4/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141
59	Dual Fuel Staged Combustion Cycle - OX Cooled, Mode I PROPELLANT: RJ-5 & LH2/Oxygen Advanced High Pressure Engine Study for Mixed-mode vehicle Appications, Aerojet Liq Roc Cmpy, NASA CR-135141
60	Staged combustion PROPELLANT: LOX/RP-1 Advanced High Pressure Engine Study For Mixed-Mode Vehicle Applications, NASA CR-135141, Aerojet Liq. Rkt. Co.
61	SSME (Extendable Nozzle) PROPELLANT: LOX/LH2 SSMS Application for Unmanned Launch Vehicle Final Briefing, USAF-SD/USAF-RPL/ROCKETDYNE 19 Febuary 1985
62	Gas Generator (case# 1 near) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
63	Gas Generator (case# 1 far) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
64	Gas Generator (case# 2) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
65	Gas Generator (case# 3 near-1) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
66	Gas Generator (case# 3 near-2) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
67	Gas Generator (case# 3 far-1) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
68	Gas Generator (case# 3 far-2) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
69	Gas Generator (case# 3A) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
70	Gas Generator (case# 4) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
71	Gas Generator (case# 5) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
72	Gas Generator (case# 6-1) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NSA8-36357, Final Review, 23/24 October 1986 Rockwell International
73	Gas Generator (case# 6-2) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
74	Gas Generator (case# 7 near) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
75	Gas Generator (case# 7 far) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
76	Gas Generator (case# 8) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
77	Gas Generator (case# 9-1) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
78	Gas Generator (case# 9-2) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
79	Gas Generator (case# 10) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
80	Gas Generator (case# 11) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
81	Gas Generator (case# 11B-1) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
82	Gas Generator (case# 11B-2) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
83	staged combustion (case# 12 near) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
84	staged combustion (case# 12 far) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
85	staged combustion (case# 13) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
86	staged combustion (case# 14) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
87	staged combustion (case# 15) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
88	staged combustion (case# 16) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
89	staged combustion (case# 17) PROPELLANT: LOX/RP-1 Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
90	staged combustion (case# 18) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
91	staged combustion (case# 19) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NAS8-36357, FINAL REVIEW, 23/24 October 1986, Rockwell International
92	staged combustion (case# 220) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
93	staged combustion (case# 21) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
94	staged combustion (case# 22) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
95	staged combustion (case# 23) PROPELLANT: LOX/C3H8 (NBP) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
96	staged combustion (case# 24 near) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
97	staged combustion (case# 24 far) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
98	staged combustion (case# 25) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
99	staged combustion (case# 26) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
100	staged combustion (case# 27) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
101	staged combustion (case# 28) PROPELLANT: LOX/C3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
102	staged combustion (case# 29) PROPELLANT: LOX/N3H8 (S/C) Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
103	staged combustion (case# 30 near) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
104	staged combustion (case# 30 far) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
105	staged combustion (case# 31) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
106	staged combustion (case# 32) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
107	staged combustion (case# 33) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
108	staged combustion (case# 34) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
109	staged combustion (case# 35) PROPELLANT: LOX/CH4 Hydrocarbon Engine Study, NAS8-36357, Final Review, 23/24 October 1986, Rockwell International
110	Mode 1 Tripropellant PROPELLANT: LOX/LH2/RP-1 Advanced Engine Study for Mixed-Mode Orbit-Transfer Vehicles, NASA CR-159491,Dec. 1978,Aerojet Liq. Rkt. Co
111	Mode 1 Dual-Expander PROPELLANT: LOX/LH2/RP-1 Advanced Engine Study for Mixed-Mode Orbit-Transfer Vehicles, NASA CR-159491,Dec. 1978,Aerojet Liq. Rkt. Co

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

Entry #	Cycle/Propellant/Source
112	Plug Cluster
	PROPELLANT: LOX/LH2/RP-1
	Advanced Engine Study for Mixed-Mode Orbit-Transfer Vehicles, NASA CR-159491,Dec. 1978,Aerojet Liq. Rkt. Co
113	Full Flow Staged Combustion (ACUREX #1)
	PROPELLANT: LOX/LH2
	ACUREX Corporation
114	FULL FLOW STAGED COMBUSTION (ACUREX #2)
	PROPELLANT: LOX/LH2
	ACRUEX CORPORATION

FIGURE 2.1-1 DATA BASE MENU OF ADVANCED/UNDEVELOPED ROCKET ENGINE CONCEPTS (CONTINUED)

• Application to NMTS
• Size/Scalability to NMTS Sizes of Interest
• Undeveloped Cycle or Advanced Concept (e.g., plug nozzle)
• Low-Cost Potential
• High-Performance Potential
• Operability Features
• Maintainability
• Near-Term/Far-Term Applicability
• Applications to Other Launch Vehicles (ALS, HLLV, etc.)
• Technology Risk/Technology Base

FIGURE 2.1-2 PRESCREENING PROCESS QUALITATIVE CRITERIA

TABLE 2.1-1 CANDIDATES FROM PRESCREENING PROCESS

1.	Staged Combustion Hybrid Cycle (LPIAG)*
2.	High/Variable Mixture Ratio Cycle (Acurex)
3.	High/Variable Mixture Ratio Cycle (Pratt and Whitney)
4.	Split Expander Cycle (STME)
5.	Split Expander Cycle (LRB)
6.	Tri-Propellant Engine Cycle
7.	Expander Bleed Cycle
8.	Full Flow Staged Combustion Cycle (Acurex)
9.	Full Flow Hybrid Cycle (Acurex)

*Liquid Propulsion Interagency Group

TABLE 2.1-2 GENERIC ROCKET ENGINE CYCLES

GAS GENERATOR CYCLES	EXPANDER CYCLES	STAGED COMBUSTION CYCLES
• Gas Generator Cycle (BASIC)	• Expander Cycle (BASIC)	• Staged Combustion Cycle (BASIC)
• Thrust Chamber Tapoff Cycle	• Dual Expander Cycle	• Full Flow Staged Combustion Cycle
• Hydrogen Bleed Cycle	• Split Expander Cycle	• Full Flow Staged Combustion/LOX Expander Cycle
• High Pressure Low Pump Discharge Cycle	• Augmented Expander Cycle	• Full Flow Staged Combustion/LH_2 Expander Cycle

2.1.1.1 Gas Generator Cycles

The gas generator cycle was the earliest pump-fed rocket engine cycle and was employed extensively as Saturn-Apollo launch vehicle engines (F-1, H-1 and J-2). A simplified flow schematic for a conventional gas generator cycle is shown in Figure 2.1-3. The gas generator cycle is known as an open cycle, as all of the engine propellants do not pass through the main combustion chamber. The schematic shows dual turbopumps using parallel flow turbine drive gases, although series flow turbine drive (Saturn engines) and a single turbine with geared fuel and oxidizer pumps (e.g., Agena) have been options used on other gas generator engines. The gas generator would normally operate very fuel-rich, to maintain a turbine inlet temperature below the thermal/structural limits of the turbine blades. The gas generator flow is used to drive the turbines which drive the fuel and oxidizer pumps. The turbine exhaust gases (in Figure 2.1-3) are then dumped into the divergent portion of the expansion nozzle at a point where the turbine exhaust gases are somewhat higher in pressure than the nozzle gases, and further expanded, producing an additional (but relatively small) amount of thrust. A separate turbine exhaust duct/nozzle has been used on some gas generator engines (e.g., Agena) instead of dumping the exhaust into the main nozzle. A secondary low chamber pressure combustion chamber, where additional oxidizer is injected to more fully combust the gases and raise the overall gas temperature, has also been proposed (i.e., dual bell concept) to regain some of the ISP lost through the low pressure/low temperature turbine exhaust dump that is inherent with open cycle engine systems. With the gas generator cycle however, the pump pressure needs only to supply system pressure drops and does not depend upon the turbopump efficiencies as closed cycle engines must. Low efficiency turbopumps however, require increased gas generator flow rates and effect the thrust chamber mixture ratio and decrease the overall ISP. The low temperature turbine exhaust gases and the shift in thrust chamber mixture ratio necessary to balance the effects on the overall mixture ratio of the fuel-rich gas

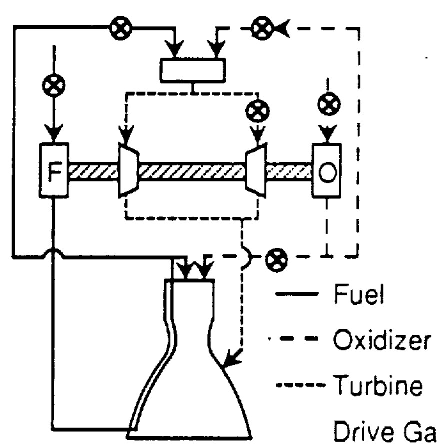


FIGURE 2.1-3 GAS GENERATOR CYCLE FLOW SCHEMATIC

generator, results in a decrease in I_{SP} of 10 to 12 seconds at a chamber pressure of 3000 psia, as compared to a staged combustion cycle at the same chamber pressure. The positive attributes of the gas generator cycle when compared to closed (topping) cycles (expander and staged combustion) include:

- Lower weights (given the same thrust, chamber pressure, etc.)
- History of high reliability
- Large experience and technology base allowing for growth with moderate technology developments

The thrust chamber tapoff cycle variation of the basic gas generator cycle is shown in Figure 2.1-4. In this cycle fuel-rich gases are tapped off the main chamber through ports in the main combustion chamber, and used to drive the turbines. This cycle was successfully tested in the J-2X engine test program (The J-2X was an experimental version of the LOX/LH_2 J-2 Saturn engine). The performance of the tapoff cycle for the same chamber pressure, mixture ratio, and nozzle area ratio, would be the same as the basic gas generator cycle if the turbine drive gases are at the same temperature and are exhausted in the same manner. The tapoff cycle could weigh less than the basic gas generator cycle because no gas generator is required. This reduction in weight through elimination of the gas generator is somewhat offset by the addition of the tapoff ports and hot gas ducting from the main thrust chamber to the turbine inlets. The tapoff cycle could have better reliability than the basic gas generator cycle, because the potential for turbine inlet temperature spikes caused by mixture ratio shifts (more oxidizer or less fuel resulting in a locally higher mixture ratio and therefore higher combustion temperature) is reduced. The technology for this cycle has been demonstrated, but the design and position of the tapoff ports being extremely critical to obtaining the proper temperature tapoff gases to drive the turbines,

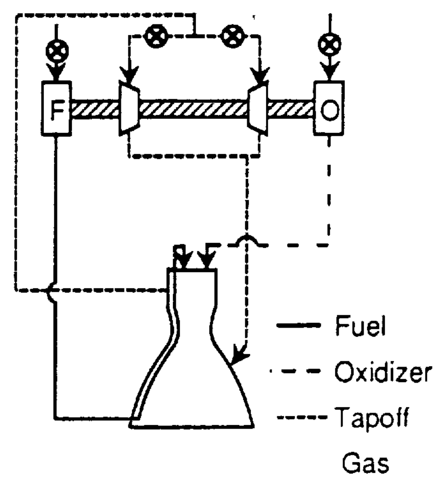


FIGURE 2.1-4 THRUST CHAMBER TAPOFF CYCLE FLOW SCHEMATIC

make the development of an engine at a higher thrust level and chamber pressure than has been demonstrated, somewhat of a technical risk.

The hydrogen bleed cycle is shown in Figure 2.1-5. In this cycle, the regenerative hydrogen coolant is used as the turbine drive gas, eliminating the requirement for a gas generator. The turbine exhaust gas (bleed flow) is dumped into the divergent portion of the main expansion nozzle. Because of the use of heated hydrogen as the turbine drive gas, the ISP is lower than that of a conventional bipropellant gas generator or topping cycle, and is applicable only to systems where high heat input to the coolant is available. This lower ISP is a result of the lower temperature of the regeneratively heated hydrogen, as compared to the temperature of the gases from a bipropellant gas generator, necessitating a large hydrogen flow requirement to drive the turbines, which results in large turbine exhaust losses. The requirement for a lower main thrust chamber mixture ratio to balance the overall mixture ratio when using hydrogen for the turbine drive gas, would further decrease ISP.

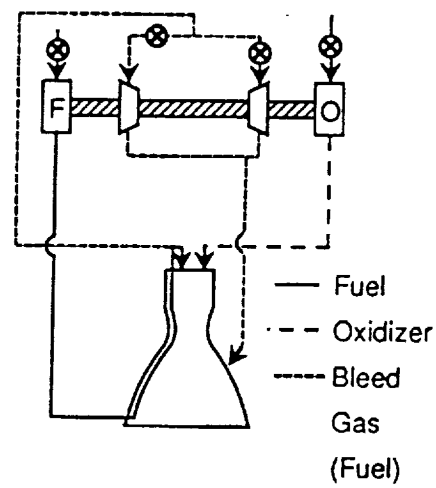


FIGURE 2.1-5 HYDROGEN BLEED CYCLE FLOW SCHEMATIC

As with the expander cycle, which will be discussed later, the pumping power potential of regeneratively heated hydrogen in the bleed cycle is probably not sufficient to attain the high thrust levels required for launch vehicle engines (300K - 500K).

The weight of a hydrogen bleed cycle engine would be comparable to the tapoff cycle engine; the engine would have no gas generator, but the weights for the turbines and hydrogen ducting would somewhat offset this advantage.

Potentially the hydrogen bleed cycle could have a very high reliability, comparable to the hydrogen expander cycle, due to the low turbine operating pressures/temperatures and the elimination of turbine inlet temperature spikes as discussed for the tapoff cycle engine, as compared to engines with gas generators.

Technology development required for the hydrogen bleed cycle would primarily be in the design of the thrust chamber coolant jacket having a high hydrogen heat load. The major issues for NMTS applications for the hydrogen bleed cycle are thrust capability coupled with ISP performance for large hydrogen bleed flows.

The high pressure low pump discharge (HPLPD) cycle is also a variation of the basic gas generator cycle as shown in Figure 2.1-6. In this variation, the combustion chamber coolant flow is minimized and used as the gas generator fuel flow, significantly reducing the pump discharge pressure requirement, since the major coolant pressure drop is in a parallel flow path rather than in series as in a conventional gas generator engine. This allows for a high chamber pressure capability with lower pump discharge pressures and turbine flows, and therefore lower turbopump weights and turbine exhaust losses. This high chamber pressure capability also permits the use of higher expansion ratio nozzles with increased sea level ISP performance. The resultant ISP increase over a conventional gas generator cycle engine would be 4 to 6 seconds at a chamber pressure of 4000 psia. With the turbopump weight reduction, the weight of a HPLPD cycle with a chamber pressure of 4000 psia would be about that of a conventional gas generation cycle operating at a 3000 psia chamber pressure. The major technology development required is the thrust chamber with its complex cooling circuit. The reliability of the HPLPD cycle should be comparable to the conventional gas generator after demonstration of thrust chamber cooling and overall design. The current STME/STBE gas generator cycle concepts are employing this cycle option at a chamber pressure of 2250 psia.

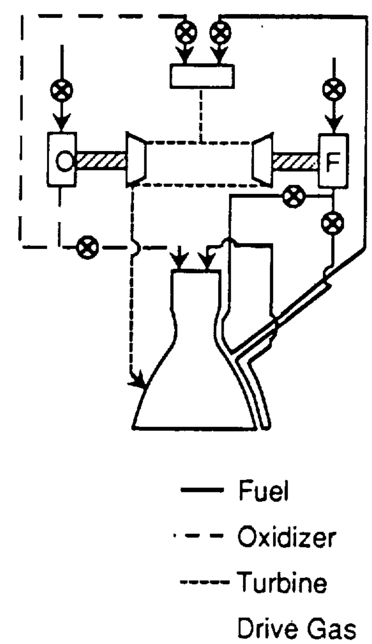


FIGURE 2.1-6 HIGH PRESSURE PUMP DISCHARGE CYCLE FLOW SCHEMATIC

2.1.1.2 Expander Cycles

Expander cycles are topping or closed cycles, because all the propellant flow goes through the main combustion chamber, unlike the gas generator cycle which is an open cycle, where part of the propellant is combusted in the gas generator and exhausted into the divergent section of the nozzle. A flow schematic for the conventional expander cycle is shown in Figure 2.1-7. In the expander cycle, as was the case in the bleed cycle, the turbine drive power comes from heat input into the thrust chamber regenerative coolant, normally the fuel. However, in the expander cycle the turbine gases are put into the main combustion chamber and not dumped into the nozzle, resulting in higher ISP when compared to the bleed cycle. The RL 10 engine used on the Centaur upper stage is an expander cycle engine using regeneratively heated hydrogen for the turbine drive. The potential ISP performance at comparable chamber pressure, mixture ratio and nozzle expansion ratio, is higher than the gas generator cycle (no turbine exhaust losses) and the same as the staged combustion cycle. However, because of the utilization of regenerative hydrogen for the turbine drive, the chamber pressure is dependent upon the heat transfer to the hydrogen coolant from the thrust chamber. Note also that all of the hydrogen flow is used to cool the thrust chamber and then drive the turbines. This results in a chamber pressure limit of about 1200 psia and more importantly a maximum thrust per engine under 100K pounds, eliminating this cycle from contention for launch vehicle applications requiring a thrust in the range of 300K - 500K pounds. An alternate version of the expander cycle called the split expander with potential capability in the 500K pound thrust range will be discussed later in this subsection.

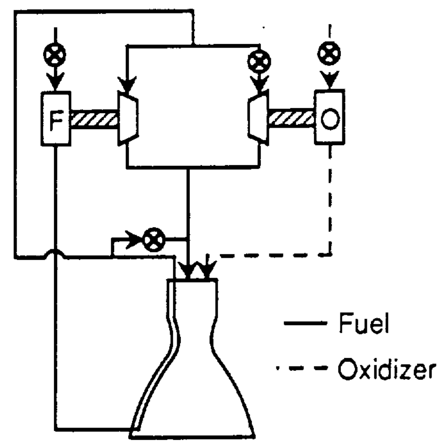


FIGURE 2.1-7 EXPANDER CYCLE FLOW SCHEMATIC

The weight of an expander cycle engine would be comparable to (but probably somewhat lighter than) a staged combustion cycle engine given equal design conditions (chamber pressure, nozzle area ratio, etc.). If a longer than normal combustion chamber is required to achieve the required heat transfer into the regenerative coolant for high chamber pressure/thrust capabilities, the result would be an increase in engine weight. Reliability of the expander cycle is higher than either the gas generator or staged combustion cycles due to:

- Elimination of hot turbine drives
- Inherent self limiting operations
- Fewer number of components (no gas generators or preburners).

Expander cycle technology development requirements for increasing chamber pressure and thrust over those of the RL 10 are considered moderate. Increasing the coolant heat loads without severely impacting overall engine weight to attain high thrust (~100K) is the most demanding technology requirement for application of the expander cycle to OTV/STV or other upper stages.

The dual expander option is an arrangement of concentric combustion chambers as shown in Figure 2.1-8. The terminology "expander" in describing this concept is actually referring to the outer chamber having an expansion-deflection (E-D) nozzle, where the plug for the E-D nozzle is the inner chamber bell nozzle. In actuality, this concept could operate in the expander drive cycle, gas generator drive cycle, or staged combustion drive cycle, depending upon other design considerations. The concentric chamber arrangement results in a lower overall area ratio (and therefore a lower ISP) but a higher total thrust performance when both chambers are operating, and a higher overall area ratio (and therefore a higher ISP), but a lower total thrust performance when only the outer chamber is operating. Another variation of the concentric chamber concept is the dual throat engine, the major differences being that in the dual throat concept, the inner chamber has a very short divergent expansion nozzle that has its exit at the throat of the outer chamber. There is no E-D nozzle for the outer chamber, only a bell. The dual throat concept operates with both chambers for a higher thrust, but at a lower overall area ratio (and ISP) and with the inner chamber only for a lower total thrust but a higher area ratio (and ISP). Like the dual expander, the dual throat concept can operate using any of the drive cycles, depending upon other design considerations. In the operation of the outer chamber only (dual expander) or inner chamber only (dual throat), some bleed flow will need to be put through the unused chamber (inner or outer respectively) to prevent recirculation of hot combustion gases back up into the engine. Regardless of whether the dual expander or dual throat concept is being considered, there are performance, weight, reliability and technology trends/issues that are

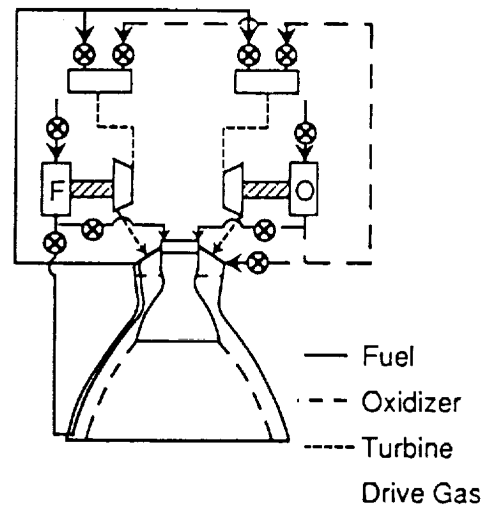


FIGURE 2.1-8 DUAL EXPANDER CYCLE FLOW SCHEMATIC

common. Both concepts have primary application to systems operating both within the atmosphere and in vacuum, such as single-stage-to-orbit (SSTO) launch vehicles. The performance advantages of dual thrust and altitude performance compensation capabilities on stage-and-a-half or two-stage vehicles, even when employing parallel engine burn, are probably not large enough to offset system weight and cost increases.

The use of either concept in the expander drive cycle is also subject to the chamber pressure and thrust limitations (<100K pounds) as discussed for the conventional expander cycle. Again the split expander drive cycle would have application to either concept, as would the gas generator and staged combustion drive cycles. The primary issues associated with these concepts are the heavy thrust chambers and injector designs that result from the complex/regenerative cooling system that is required because of the concentric thrust chamber configuration. In addition, there are a large number of control valves required to operate these concepts with resulting increases in weight and decreases in engine reliability. Ultimately the reliability of these concepts will depend on the technology development of robust concentric combustion chambers and methods to cool them.

The split expander cycle is a rescent derivation of the conventional expander cycle for attaining launch vehicle engine size thrust levels (~500K pounds). As shown in the flow schematic of Figure 2.1-9, only a portion of the fuel is used for regenerative cooling of the thrust chamber and driving the turbines, with the remainder of the liquid fuel being injected directly into the main combustor. This reduces the specific pump power requirement and increases the maximum thrust level capability of the drive cycle, such that a 500K pound engine has been shown analytically to be feasible, and has been proposed for the STME. However, the chamber pressure capability (<1500 psia) is well below that of gas generator and staged combustion cycles. For comparable chamber pressures, mixture ratios and area ratios, the ISP of the split expander is comparable to the staged combustion

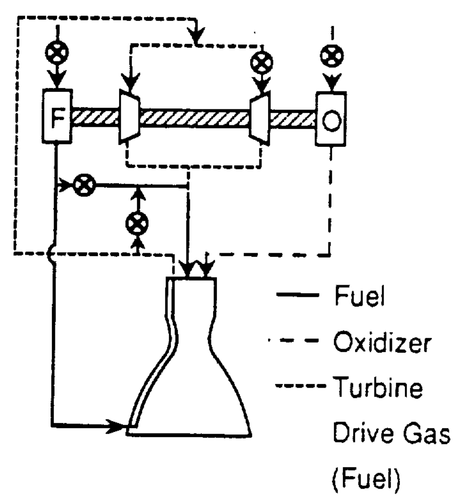


FIGURE 2.1-9 SPLIT EXPANDER CYCLE FLOW SCHEMATIC

cycle and higher than the gas generator cycle (the higher performance of STME gas generator cycle engine concepts as compared to split expander cycle concepts is attained through higher chamber pressures and higher expansion ratio nozzles).

For comparable design conditions the split expander could weigh less than either the staged combustion or gas generator cycle alternatives, depending upon the combustion chamber length required to achieve the necessary heat transfer into the regenerative coolant for turbine drive. The reliability of the split expander cycle is comparable to the conventional expander cycle, which could be higher than the gas generator or staged combustion cycles, because it eliminates very hot turbine drives and has fewer major components (no gas generators or preburners). The technology development requirements for the split expander include:

- Demonstration of the thrust chamber heat loads and cooling methodology for high thrust high chamber pressure engines
- Repeatibility of the engine start transient.

The augmented expander cycle, as shown in Figure 2.1-10, is a topping (closed) cycle that uses an expander cycle to drive the fuel turbine and the staged combustion cycle to drive the oxidizer turbine. This cycle is very similar to the hybrid staged combustion cycles to be discussed in Subsection 2.1.1.3. In the augmented expander cycle, the preburner is used to augment pump power to attain higher chamber pressures and thrusts than would be possible in a conventional expander cycle. For the same design conditions (i.e., chamber pressures, mixture ratio, and nozzle area ratio) the augmented expander cycle would nominally have an ISP comparable to the expander and staged combustion cycles. The relative weight for the augmented expander for comparable thrust levels would be heavier than the conventional expander cycle, but somewhat lighter than the staged combustion cycle. In terms of reliability, the single preburner of the augmented expander cycle could decrease reliability as compared with the

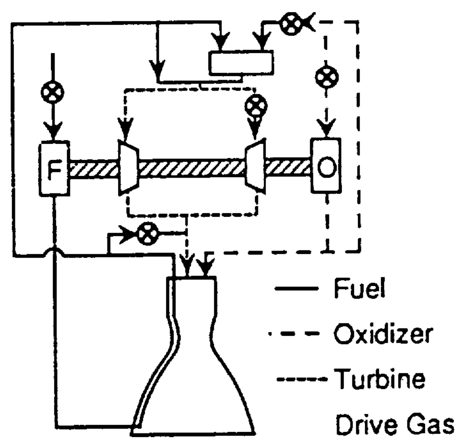


FIGURE 2.1-10 AUGMENTED EXPANDER CYCLE FLOW SCHEMATIC

conventional expander cycle, but as compared with the staged combustion cycle, because of the lower turbine temperatures, decreases the possibility of high temperature spikes.

The major issue involved with the utilization of this cycle for NMTS is the size of the preburner required to attain a thrust level of approximately 500K pounds. The major technology development for this cycle is the integration of the small preburner in an expander cycle engine. This concept can also be thought of as a hybrid staged combustion cycle and additional discussion will be offered at the end of Subsection 2.1.1.3.

2.1.1.3 Staged Combustion Cycles

Staged combustion cycles are closed (topping) cycles where all of the engine propellants flow through the main combustion chamber (Figure 2.1-11). The pre-combustors or preburners generate the drive gases for the turbines as gas generators do for the gas generator cycles. However, unlike the gas generator cycle, the fuel-rich turbine exhaust is routed to the main combustion chamber where it is combusted with additional oxidizer, eliminating the turbine exhaust ISP losses of the gas generator cycle. Configurations for staged combustion cycles could have single or dual preburners. (Figure 2.1.11 shows a single preburner configuration.) The SSME is a LOX/LH_2 staged combustion cycle with dual fuel-rich preburners, one for the LOX turbopump drive and one for the LH_2 turbopump drive. Other drive gas options are also possible, such as fuel-rich on the fuel side and oxidizer-rich on the oxidizer side, as will be discussed later for the full flow staged combustion cycle. The result of having a topping cycle with combustion gas driven turbines is the capability of having a high chamber pressure/high thrust engine with a high ISP, which couples the best performance features of the convention gas generator and expander cycles. However, this performance requires high technology, complex turbopumps to attain the high efficiency and high pump discharge pressures necessary to meet all the system pressure drop requirements for operating the main chamber at high pressure. This high chamber pressure allows for larger nozzle area

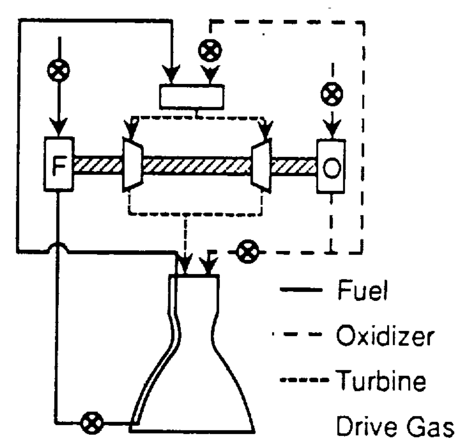


FIGURE 2.1-11 STAGED COMBUSTION CYCLE FLOW SCHEMATIC

ratios and higher ISP as compared to the gas generator and expander cycles. Even though thrust chamber weight deceases with increased chamber pressure, any overall engine weight decrease is offset by the rapidly increasing turbopump and powerhead weights resulting from the increased pump power required for increased chamber pressure as compared to high chamber pressure gas generator cycle concepts.

Tripropellant and dual fuel configuration options are possible using the staged combustion cycle, as they are with the gas generator and expander cycles, depending on design requirements.

Key reliability issues with the staged combustion cycle are the avoidance of the high turbine temperature spikes associated with the start transient and the structural and thermal design of the turbopumps with particular attention to the bearings and their lubrication and cooling. The SSME has given the staged combustion cycle a strong technology base. Current SSME improvement efforts are focusing on lower manufacturing, operations and maintenance costs and increased reliability and lifetime for the LOX turbopump.

The full flow staged combustion (FFSC) cycle (Figure 2.1-12) could be adaptable to various propellant combinations (LOX/LH_2 is shown). This rocket cycle uses fuel-rich combustion products to drive the fuel turbine and oxidizer-rich combustion products to drive the oxidizer turbine. All propellant flow is ultimately burned in the main chamber (topping cycle) and exhausted through the main chamber throat giving this cycle maximum ISP.

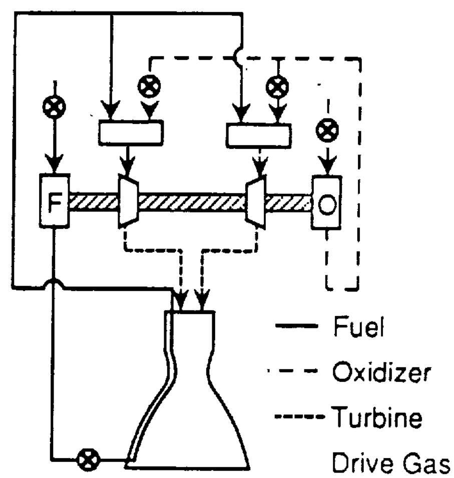


FIGURE 2.1-12 FULL FLOW STAGED COMBUSTION CYCLE FLOW SCHEMATIC

All propellant flow is used to drive the turbines resulting in the lowest possible turbine inlet temperature for a given main chamber pressure. In addition, the utilization of all propellants as turbine drive gases allows for gas-gas main chamber injection with potential mixing and combustion efficiency increases as compared to the current SSME gas-liquid injection scheme. Conceptual design studies of engines based on this cycle performed by the Acurex Corporation have shown such engines to be of noteworthy simplicity, with the potential of offering low cost without compromises in safety, reliability, weight, or performance as compared to the SSME staged combustion cycle option. Because of the low LOX turbine temperatures, LOX turbopump bearing thermal/structural stresses will be decreased over those encountered in the current SSME, and overall LOX turbopump life could be extended as compared to current SSME experience. However, LOX-rich combustion gases for turbine drives present other technology problems which must be solved with technology development projects to resolve concerns.

Hybrid full-flow staged-combustion cycles (i.e., all propellants used for turbine drive) would employ the staged combustion cycle for one propellant turbopump drive and the expander cycle for the other propellant turbopump drive. A full flow staged combustion/LOX expander cycle is shown in Figure 2.1-13. In this full flow cycle the oxidizer-rich preburner is eliminated and an oxidizer heated in a regenerative cooling jacket on the main combustion chamber is substituted to drive the LOX turbine.

This cycle may be simpler in some respects than the FFSC cycle, because it only has a single fuel-rich, preburner, and has no LOX-rich combustion gas turbine drive. However, problems associated with warm LOX for turbine drive, as well as LOX regenerative chamber cooling must be resolved in technology development projects. In addition, this cycle is limited in chamber pressure to on the order of 2000 psia by the local cooling capability of the oxidizer, and is heat transfer limited on engine size and may not have thrust capabilities over 450K. It is also not particularly well suited to high or variable mixture ratio operations.

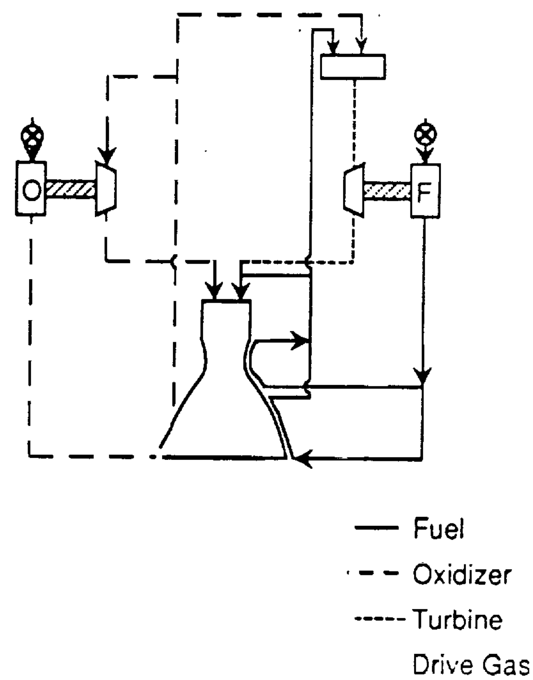


FIGURE 2.1-13 FFSC/LOX EXPANDER CYCLE FLOW SCHEMATIC

The other hybrid full flow staged combustion cycle being considered would operate in the staged combustion cycle on the oxidizer side, and the expander cycle on the fuel side. This is essentially the same as the augment expander cycle shown in Figure 2.1-10. An oxidizer-rich preburner is used for the LOX turbine drive, and LH_2 is used to cool the main thrust chamber, with the resulting heated hydrogen used to drive the LH_2 turbine.

Having only one preburner, it may be perceived by some to be simplier than the FFSC baseline cycle, and it adapts more readily to variable mixture ratio operation than the LOX expander hybrid cycle. However, because of the low LH_2 flow rate (one seventh of the engine total at a mixture ratio of 6) and the relatively high power demand of the LH_2 pumping system, this cycle is limited to chamber pressures on the order of 1200 psia and thrusts of less than 300K.

2.1.1.4 Generic Rocket Cycle Assessment Summary

The cycle options from the three generic cycle groups, gas generator, expander, and staged combustion were assessed to determine the most promising candidates for further detailed definition in Subtask 2. Criteria for this assessment included:

- Applicable to the main propulsion system of NMTS vehicles (STS Evolution, PLS/CRV, and AMLS) by having the capability of thrust levels on the order of 500K pounds using LOX/LH_2 propellants.
- Undeveloped cycle concepts, as opposed to innovative nozzle concepts (e.g., plug, aerospike) and tripropellant/dual fuel concepts that could be used with any cycle.
- Undeveloped cycles that are not, nor have not been the subject of extensive past studies and engineering demonstrations, such as the expander cycle (RL 10), gas generator cycle (Saturn engines, STME, STBE), gas generator tapoff cycle (J-2X), fuel-rich, dual preburner staged combustion cycle (SSME).
- Cycles that show potential for reducing costs, enhancing reliability/maintainability, and increasing (or not severely degrading) performance.

Subject to these criteria we made the following assessment. The pure gas generator cycle has previously been developed (F-1, J-2, H-1, etc.) but not at high chamber pressures (>1500 psia). However, the STME baseline is currently a gas generator engine with a chamber pressure of 2250 psia. The thrust chamber tap-off cycle has been the subject of a development program in the J-2X experimental engine previously tested. The hydrogen bleed cycle probably does not have the pumping power for 500K pounds thrust, and the turbine exhaust losses (all hydrogen gas) severely impact its I_{SP}. The high pressure, low pump discharge cycle is applicable to NMTS and has high chamber pressure and I_{SP} capability, but is being addressed in the STME program. For these reasons we did not recommend any gas generator concept for further definition.

The conventional expander cycle lacks power for 500K thrust level capability and has been previously developed (RL10). The dual expander, if operating in the conventional expander cycle, also lacks power for producing thrusts at the 500K level. The dual expander and dual throat concepts could also be used with other cycles, but these have primary application to SSTO vehicles, which are not current NMTS vehicle candidates. The augmented expander cycle is similar in concept to the hybrid staged-combustion cycle and does not have the 500K thrust capability necessary for NMTS applications. The split expander cycle is applicable to NMTS and shows cost reduction, enhanced reliability/maintainability potential, and could be a candidate for further definition in Subtask 2.

The SSME version (dual fuel-rich preburners) of the staged combustion cycle is developed. The dual preburner (fuel-rich on fuel side and LOX-rich on LOX side) full flow cycle appears to have potential for lower costs and enhanced reliability and maintainability. This cycle also could have high combustion efficiency because of the gas/gas injection inherent in the full flow concept. The cycle also has high/variable mixture ratio performance potential and could be a candidate for further definition in Subtask 2. The hybrid full flow staged combustion/LOX expander cycle (staged combustion on fuel side, expander on LOX side) could also have some of the same potential without having a LOX-rich combustion preburner, but would have a requirement for LOX regenerative cooling. It could also be a candidate for further definition in Subtask 2.

2.1.2 Identification and Screening Results

Data from the nine cycle concepts of Table 2.1-1 and the generic cycle assessment were further evaluated to determine the cycles to be selected for further definition in Subtask 2. The two major criteria in evaluating the concepts from Table 2.1-1 were: 1) the uniqueness of the concept, i.e., multiple versions of the same undeveloped cycle were included in the results of the prescreening, and 2) the availability of data on the concept/cycle.

The final results of the screening were the selection of the Split Expander Cycle and Full Flow Staged Combustion Cycle (including data on the LOX Expander Hybrid option) for further definition in Subtask 2. A generalized set of data will be defined for these selected undeveloped cycles in Subtask 2. No one specific contractor/study data will be used, for the split expander cycle. However, Acurex, a subcontractor to SRS on this task, formulated the parametric performance data for the full flow staged-combustion cycles.

2.2 Detailed Engine Descriptions (Subtask 2)

The search for better rocket engine cycles is driven by the need to reduce the costs of space launch operations. The aerospace community has historically been accustomed to designing and working at the leading edge of technology where unknowns predominate, and risks are taken for the payoff of higher system performance with cost not being a primary issue.

As a result of this traditional approach, rocket engines have been designed to the limits of turbine material temperatures and stresses, and combustion chamber heat transfer and cooling. Chamber pressures have been increased as high as possible and highly complex control systems have evolved. Each element of the design has been pushed to a limit or constraint, in the belief that a design at the thermal or structural limit (or some other constraint) is inherently best and will result in the most cost-effective system.

In many such instances the design constrain instead translate directly to higher cost and/or to costly development programs. For example, meeting a turbine temperature limit, may require a reduced injector pressure drop which in turn may reduce the combustion stability margin, necessitating an injector baffle development program. Rocket engine developers are well acquainted with the myriads of such tradeoffs and decisions that accompany most rocket engine programs.

Often overlooked in many of engine programs is that limitations and programs arise as a result of the initial cycle selection and/or the operating range chosen for a particular engine cycle. As an example, gas generator cycles with overboard bleed of turbine exhaust place a significant emphasis on thermal and fluid efficiency of the system to minimize ISP loss from the turbine drive flow. There is incentive to run the turbine(s) as hot as possible with attendant problems with materials, and the need for high tip speed (high centrifugal stresses) to best recover the energy from the hot gases. This requires the use of the best possible material (interpreted as highest cost quality) machined to highly precise geometry (interpreted as highest cost machining), based upon elaborate aerodynamic and thermal structural analyses (interpreted as highest cost analyses).

There is also incentive to achieve the highest possible fluid efficiency. This incentive leads to thin, highly stressed, smooth blading of precisely defined contour. In a route analogous to that described above, this incentive leads to high cost equipment based on high cost analyses and designs that require high-cost development programs to produce a reliable system.

Designs can focus on those which can be demonstrated with confidence in a more modest cost demonstration program, and expensive development programs can potentially be avoided.

The point of the foregoing discussion is to illustrate that decisions at the outset of a program are crucial in distinguishing between a low cost route and a high cost route. Once selection of a high performance, high pressure cycle is made, the drive towards high costs has begun. And experience has shown these incentives prevail over the desires for low cost.

The two selected undeveloped cycles, the split expander and full flow staged-combustion are characterized and parameterized to provide detailed engine data for engine and vehicle trade studies and assessments.

The split expander data was generated by the SEEM (Split Expander Engine Model), an SRS modification to the rocket engine preliminary design models provided by Mr. Robert Champion (NASA-MSFC/PD13), which were developed in the mid-1970's for advanced staged-combustion (ASCEM-Advanced Staged Combustion Engine Model) and gas generator (AGGEM - Advanced Gas Generator Model) cycle engines. The SEEM code is based upon the earlier codes, but has been adjusted to reflect the split expander cycle (e.g., no preburners) and in addition has the capability to evaluate turbopump LH_2 bypass flow, turbine inlet temperature, and pump speed requirements as a function of chamber pressure, vacuum thrust, mixture ratio, etc. The code gives engine performance, geometry and weight estimates very similar to those seen for split expander cycle engines in the STME/STEP programs.

The Acurex Corp. initially supplied engine performance, geometry and weight data for the full flow engine cycles. However, SRS has subsequently developed an engine model for the full flow staged-combustion cycle (FFSCEM), based on the previously developed ASCEM code, to equalize the basis for comparison of engine performance, geometry, and weight estimates.

2.2.1 Split Expander Cycle

NASA's underlying need for a low cost, dependable, reliable, and rugged rocket engine has led to many studies of various rocket engine cycles. The three STEP contractors, Aerojet Techsystems, Pratt & Whitney, and Rocketdyne, are currently studying the split expander cycle as part of the Advanced Launch System (ALS) program. This cycle is a derivative of the expander cycle, which incidentally has a successful performance record. The requirements of the split expander engine study call for a relatively high thrust engine, the ~580Klb range, as compared to the lower (16Klb) range of current expander cycle engines. The split in the fuel flow makes the higher thrust levels possible by only sending part of the fuel through the nozzle jacket. By passing the fuel through the nozzle jacket, two processes that are advantageous to engine performance take place. One, the nozzle is regeneratively cooled by the liquid fuel and two, the fuel picks up the sensible heat in the nozzle and expands to a gas then drives the fuel and LOX turbopumps. Eventually all of the fuel is mixed with the LOX and all propellants pass through the main chamber making this a topping cycle. A summary of some characteristics and issues pertaining to this relatively simple cycle include:

CHARACTERISTICS/FEATURES/OPTIONS

- Topping cycle (all propellant thru main chamber versus GG where part is dumped overboard)
- Turbine drive power from heat input into fuel regenerative coolant (self limiting)
- Only a portion of the fuel is expanded for turbine drive. The remainder is flowed directly to injector from first stage of LH_2 turbopump
- Proposed for STME/STBE concepts
- Reduces pumping power requirement and increases thrust level capability

PERFORMANCE

- For comparable chamber pressure and area ratio, same performance as the staged combustion cycle
- Chamber pressures are dependent upon the heat transfer to the hydrogen coolant from the chamber (maximum chamber pressure less than staged-combustion cycle)
- High thrust levels possible with split expander cycle over conventional expander cycle

WEIGHT COMPARISON

- Similar to the staged combustion for equal design conditions
- May require long combustion chamber to achieve heat transfer resulting in increased weight which may be offset somewhat by using lower weight A1 pump materials

RELIABILITY

- Eliminates hot turbine drives
- Emphasis on thrust chamber heat transfer
- All operational modes must be evaluated
- Has higher reliability based on the number of components (vs. staged combustion)

TECHNOLOGY

- Demonstration of thrust chamber heat loads for high thrust engines
- Repeatibility of engine start is critically dependent on engine thermal conditions

The STEP Phase B reference design requirements, as stated by NASA as of December 21, 1989 are shown in Figure 2.2-1. These provide the parametric data ranges necessary to establish the engine parametrics.

Split Expander Reference Engine (STEP Baseline Alternate)

Parameter	Design Point
Thrust:	580,000 LBF-vac
Mixture Ratio:	6.0:1
Area Ratio:	20:1
Chamber Pressure:	1100 psia
Isp:	424 sec-vac

Split Expander Alternate Engine

Parameter	Design Point
Thrust:	600,000 LBF-vac
Mixture Ratio:	6.0:1
Area Ratio:	18:1
Chamber Pressure:	900 psia
Isp:	418 sec-vac

BOTH ENGINE CYCLES

Fuel Inlet Rating Conditions:	35 PSIA @ 37.5R
Oxidizer Inlet Rating Conditions:	130 PSIA @ 166R
Minimum Fuel Inlet Conditions:	30 PSIA @ 37 R
Minimum Oxidizer Inlet Conditions:	47 PSIA @ 164 R
Maximum Fuel Inlet Conditions:	38 PSIA @ 38 R
Maximum Oxidizer Inlet Conditions:	285 PSIA @ 167 R
Design Mission Life:	10 Equiv Duty Cycles

CONFIGURATION FEATURES

- Expendable
- No Boost Pumps
- No Bleeds
- Fixed Thrust
- Open Loop Control
- Common Core/Booster Nozzle
- No Pogo Accumulator (Will Be On Vehicle)
- 10 Degree Gimbal Capability (Square Pattern)
- No Scissors Ducts (Integrated Flexible Feed System on Vehicle)
- 2 Duct Diameters Length of Straight Duct Upstream of Pump Inlets

OPTIONAL FEATURES

- Dual Thrust (Single Step)
- Ocean Recovery (Environ. Ranges from Light Sea Water Exposure to Immersion)
- Bleed System
- Dual Nozzles

FIGURE 2.2-1 STEP SPLIT EXPANDER CYCLE REFERENCE DESIGN REQUIREMENTS

In addition to the split expander studies to be performed under the .ALS program, a contracted effort through LeRC to design and build a demonstration model of a split expander engine with thrust levels up to 50K pounds will be performed by Pratt and Whitney.

2.2.1.1 Cycle Schematic

Figure 2.2-2 is a schematic of the split expander cycle. Assuming the likelihood of engine startup with tank head pressure, the jacket bypass value and the turbine bypass valve must be initially closed. This allows the fuel to pass through the nozzle and pick up enough sensible heat to drive the turbopumps. It is believed that the amount of heat picked up by the fuel will be sufficient for the initial startup period (~3 sec). An initial chill down of the turbopumps to prevent cavitation may be necessary. A spark plug ignitor will light the LOX-fuel (fuel rich mixture). The jacket bypass value and the turbine bypass valve will gradually be open as buildup occurs. With the engine operating at/or near full power, the fuel flow is split, with the majority of the fuel flow going directly to the chamber along with all of the LOX flow. The connection between the vehicle side of the gimbal and the engine side will be accomplished by an integrated flexible feed system on the vehicle. At one time, a scissors type bellows gimbal was planned but was later scraped. Ten degree engine gimbal limit requirements along with improved gimbal technology should support the choice of an integrated flexible feed system. Heat exchangers and regulators may also be required to provide LH2 and LOX tank pressurization gasses.

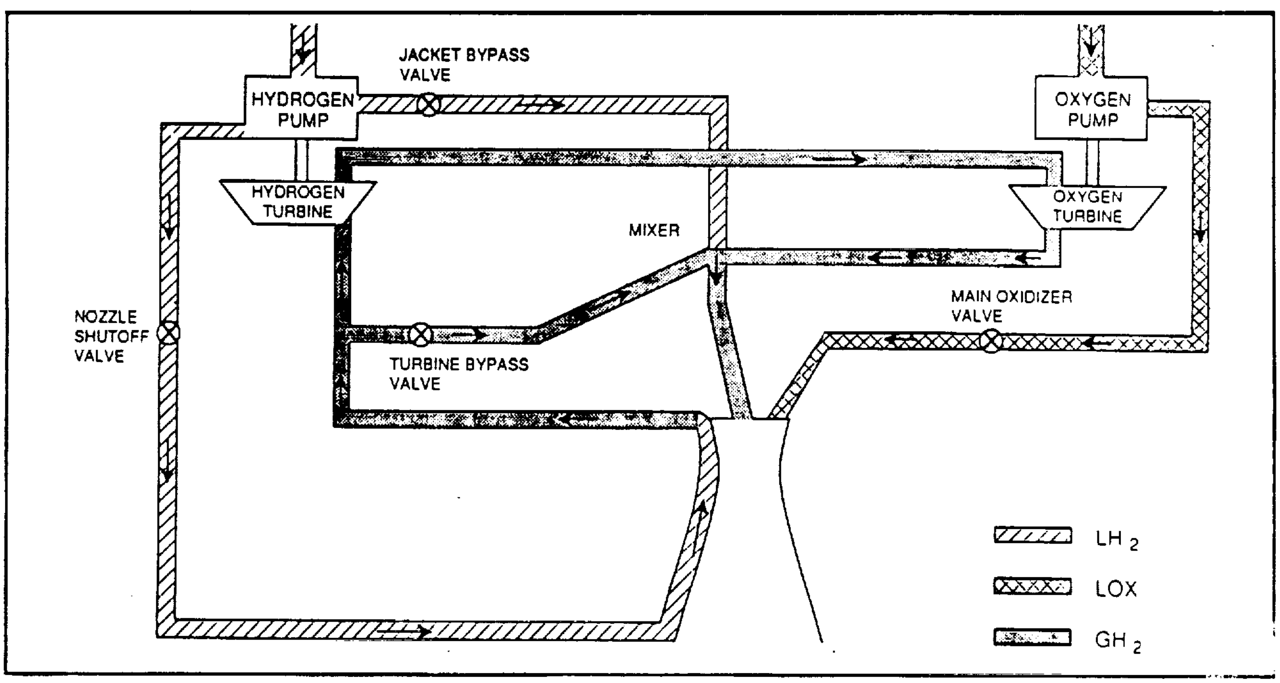
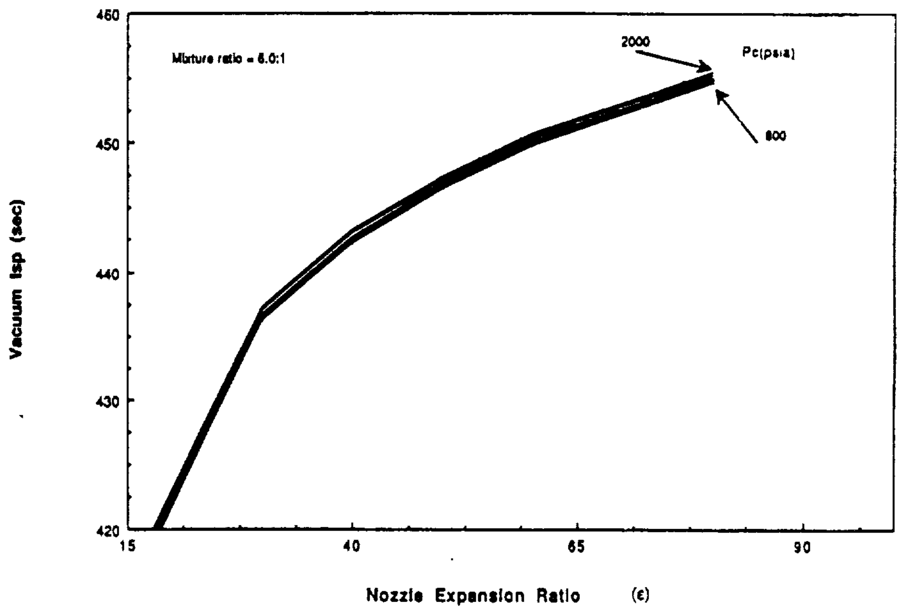


FIGURE 2.2-2 SPLIT EXPANDER CYCLE SCHEMATIC

2.2.1.2 Parametric Data

The split expander cycle has the potential to have a higher Isp for a given chamber pressure, expansion ratio, and expansion ratio, than open LOX/LH2 cycles, such as the gas generator cycle, because overboard bleed losses are avoided. Vacuum Isp versus nozzle expansion ratio (ε) and chamber pressure (Pc) for a mixture ratio of 6.0:1 and Vacuum Isp versus mixture ratio and nozzle expansion ratio at a chamber pressure of 1200 psia are shown in Figure 2.2-3.



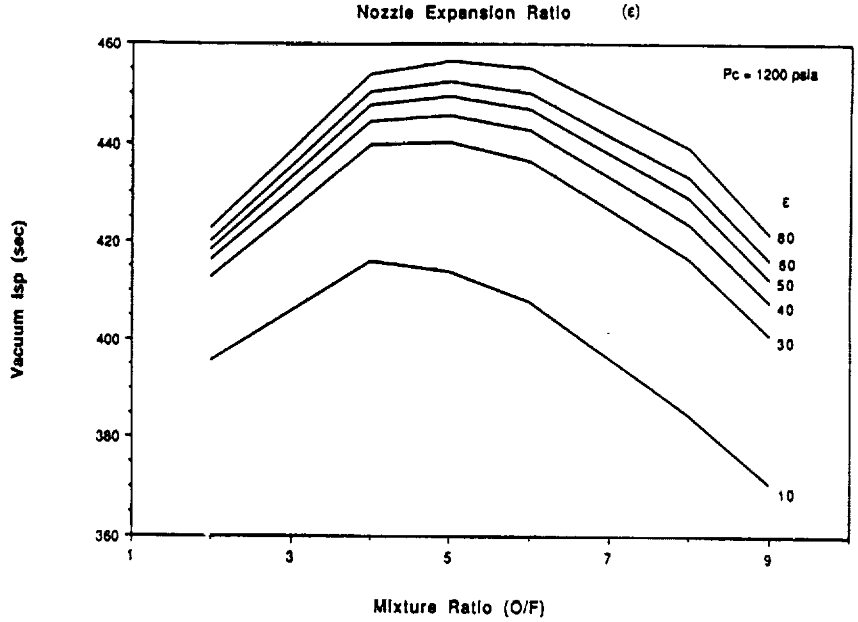
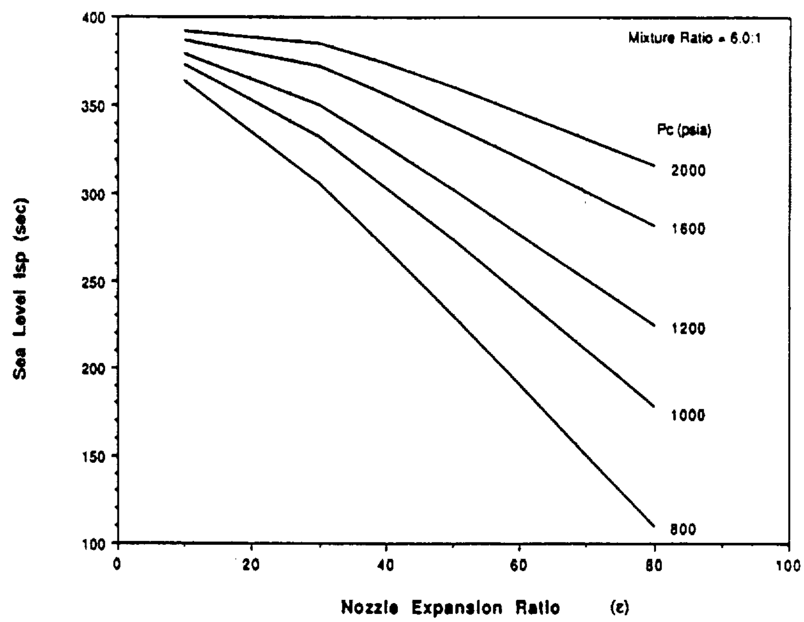


FIGURE 2.2-3 VACUUM Isp VERSUS NOZZLE EXPANSION RATIO AND CHAMBER PRESSURE AND VACUUM Isp VERSUS MIXTURE RATIO AND EXPANSION RATIO

Sea level ISP versus nozzle expansion ratio and chamber pressure at a mixture ratio of 6.0:1 and Seal level ISP versus mixture ratio and nozzle expansion ratio at a chamber pressure of 1200 psia is shown in Figure 2.2.-4.



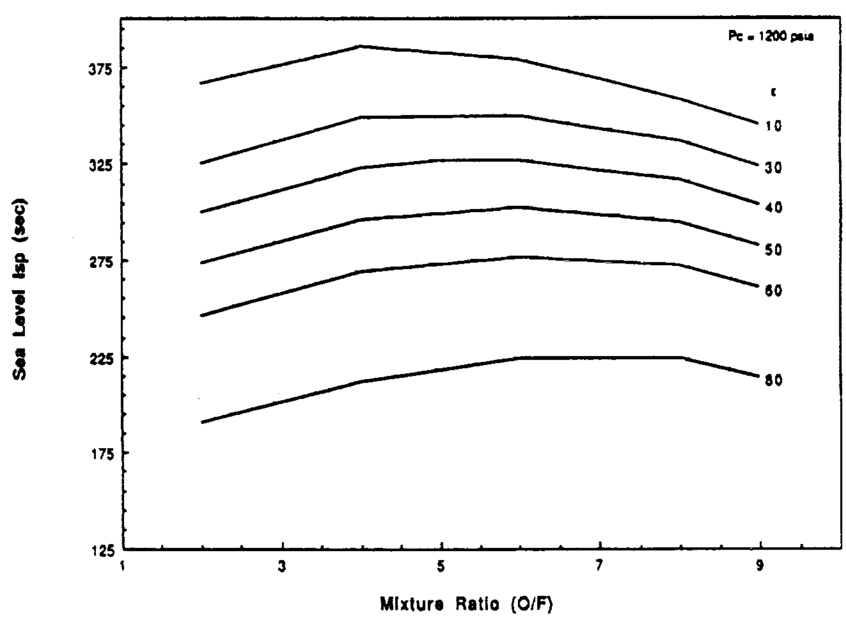


FIGURE 2.2-4 SEA LEVEL ISP VERSUS NOZZLE EXPANSION RATIO AND CHAMBER PRESSURE AND SEAL LEVEL ISP VERSUS MIXTURE RATIO AND EXPANSION RATIO

The split expander cycle, as shown in the cycle schematic (Figure 2.2-1) bypasses some of the LH_2 for chamber cooling. The heated LH_2 is then used to power the LH_2 and LOX turbopumps. For various assumptions on thrust chamber materials and cooling passage design and construction, the maximum achievable chamber pressure for the split expander cycle engine as a function of thrust level was calculated and is shown in Figure 2.2-5.

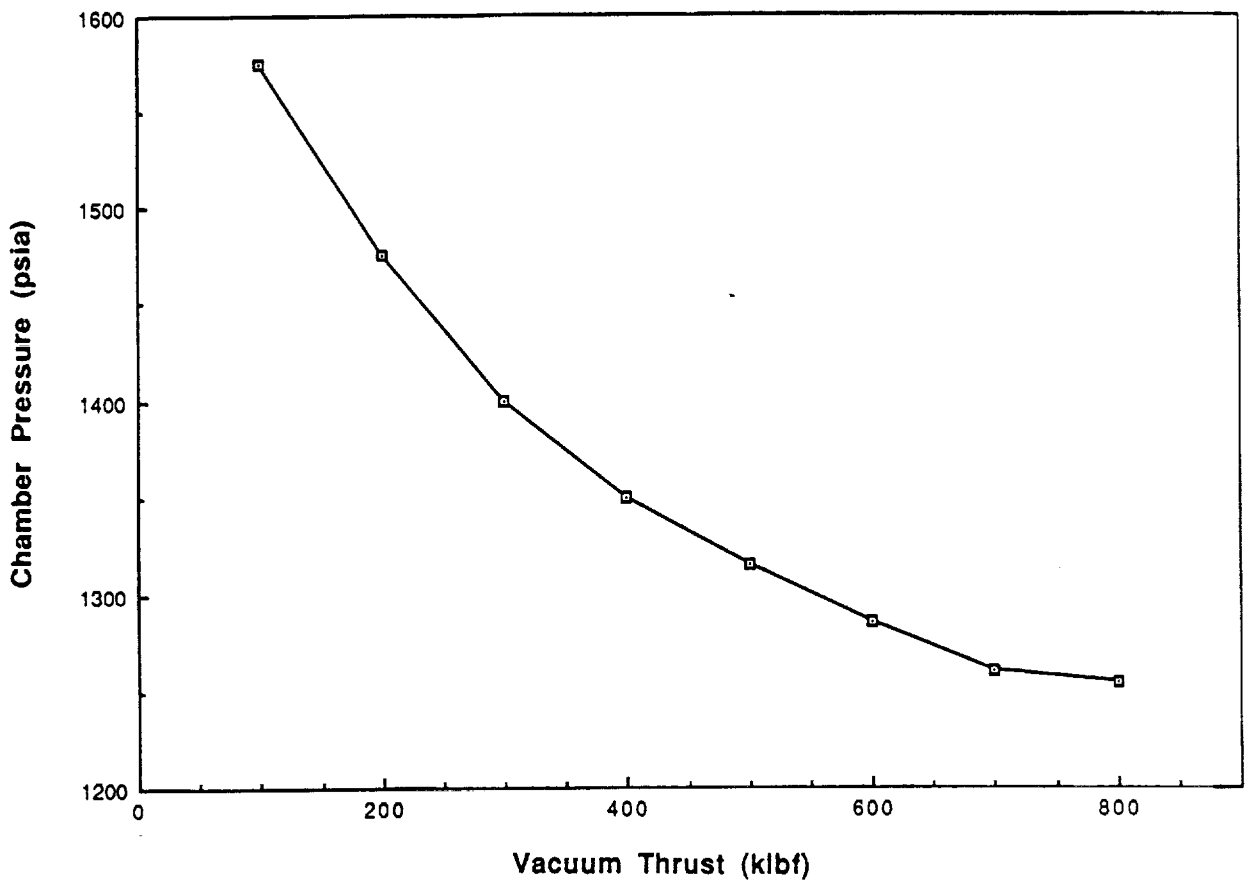
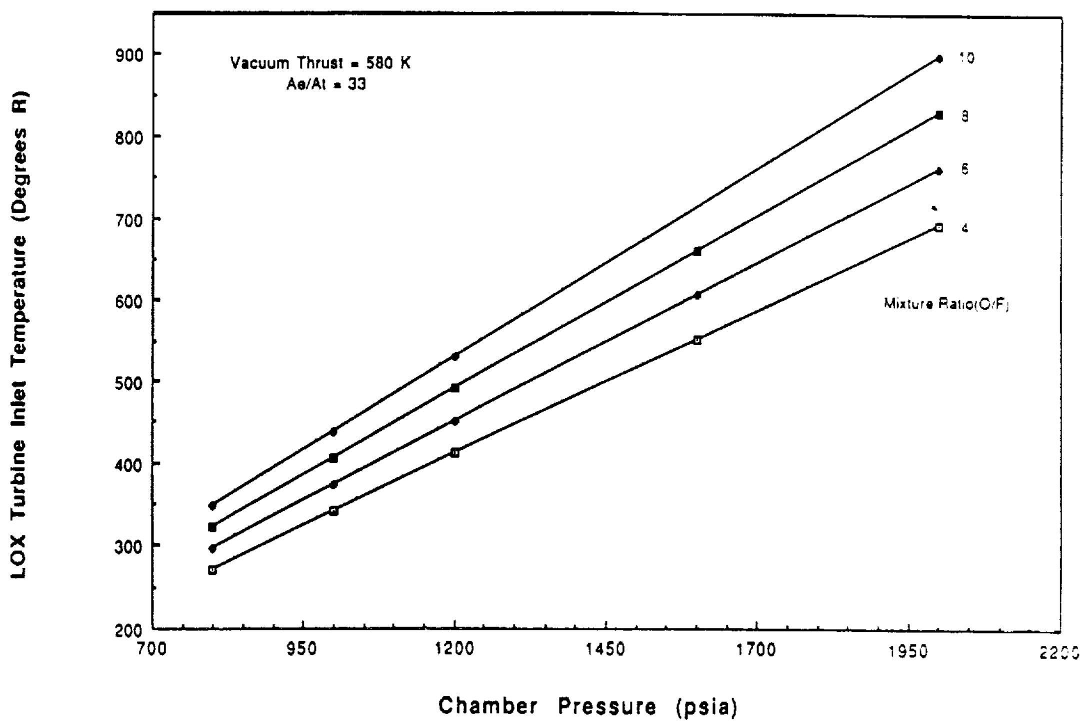


FIGURE 2.2-5 SPLIT EXPANDER CYCLE MAXIMUM ACHIEVABLE CHAMBER PRESSURE AS A FUNCTION OF THRUST

The heat flux generated by the combustion chamber of a LOX/LH_2 engine is dependent upon the combustion gas temperature (a function of mixture ratio) and the size of the thrust chamber (a function of thrust level and chamber pressure). Fuel (LH_2) and oxidizer (LOX) turbine inlet temperatures as a function of chamber pressure and mixture ratio for a fixed vacuum thrust of 580K pounds are shown in Figure 2.2-6.

LOX Turbine Inlet Temperature vs. Chamber Pressure
at Mixture Ratios



Fuel Turbine Inlet Temperature vs. Chamber Pressure
at Mixture Ratios

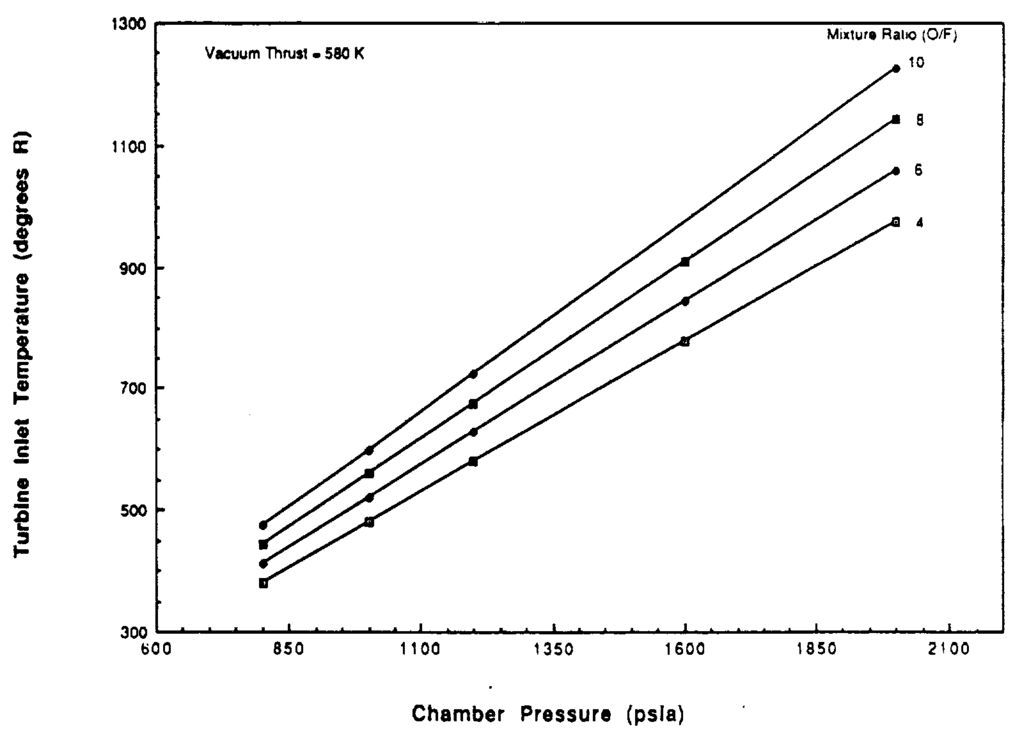
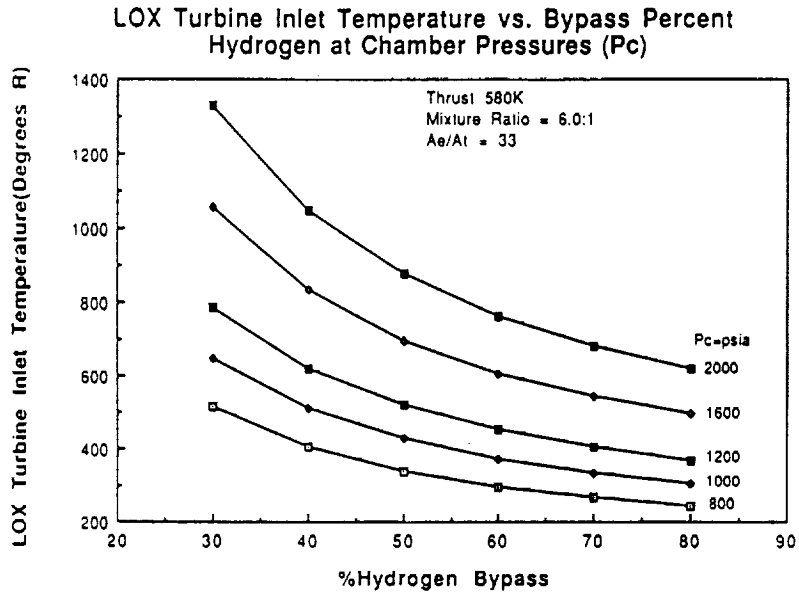


FIGURE 2.2-6 FUEL AND OXIDIZER TURBINE INLET TEMPERATURES VERSUS CHAMBER PRESSURE AND MIXTURE RATIO

The variation in turbine inlet temperature as a function of the amount of LH_2 bypassed the chamber cooling and use as the turbopump drive fluid is shown in Figure 2.2-7 as a function of chamber pressure for both the fuel (LH_2) and oxidizer (LOX) turbines.



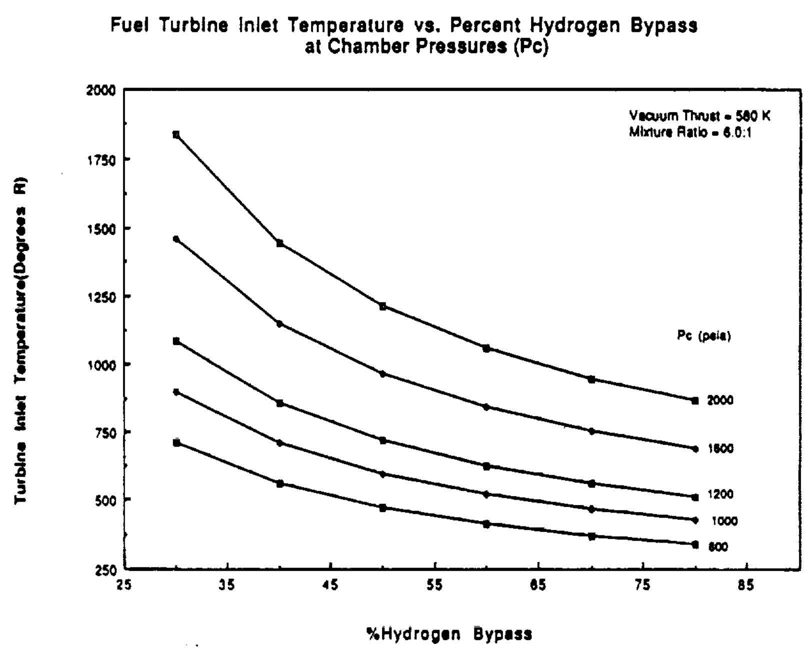
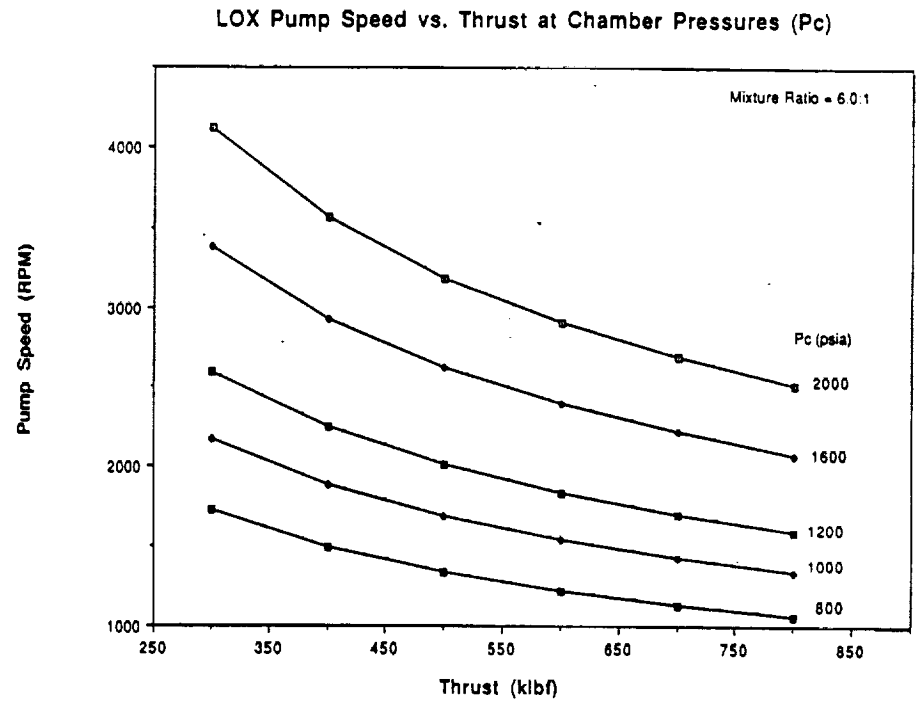


FIGURE 2.2-7 FUEL AND OXIDIZER TURBINE INLET TEMPERATURE REQUIREMENTS AS A FUNCTION OF HYDROGEN BYPASS AND CHAMBER PRESSURE

Fuel and oxidizer pump speeds as a function of thrust and chamber pressure (at a mixture ratio of 6.0:1) are shown in Figure 2.2-8.



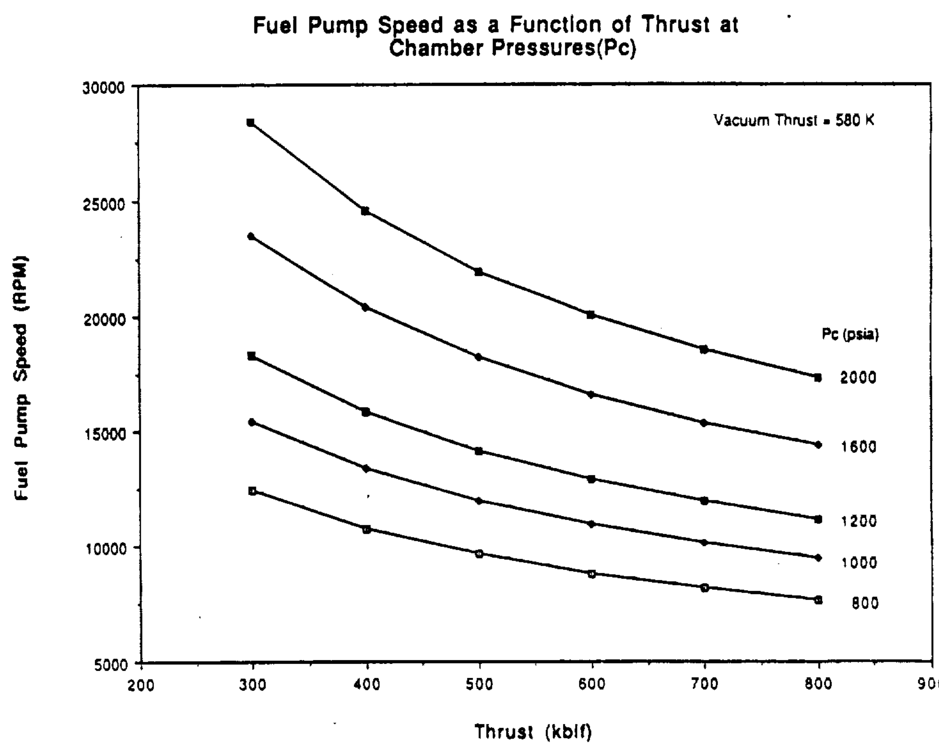
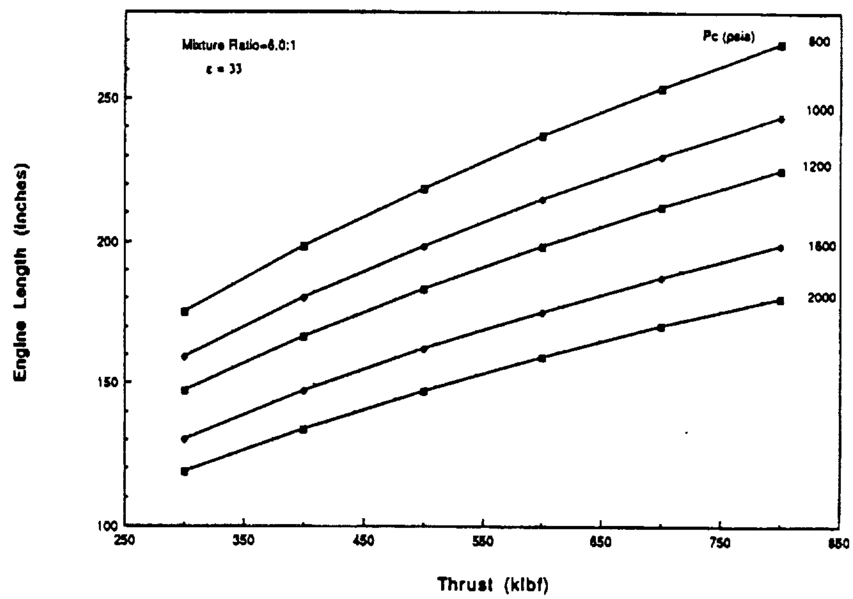


FIGURE 2.2-8 FUEL AND OXIDIZER PUMP SPEEDS AS A FUNCTION OF THRUST AND CHAMBER PRESSURE

Engine dimension data are shown in Figure 2.2-9 which gives respectively engine length versus vacuum thrust and chamber pressure (at a mixture ratio of 6.0:1 and a nozzle expansion ratio of 33:1) and nozzle exit area as a function of vacuum thrust and chamber pressure at the same mixture ratio and expansion ratio.



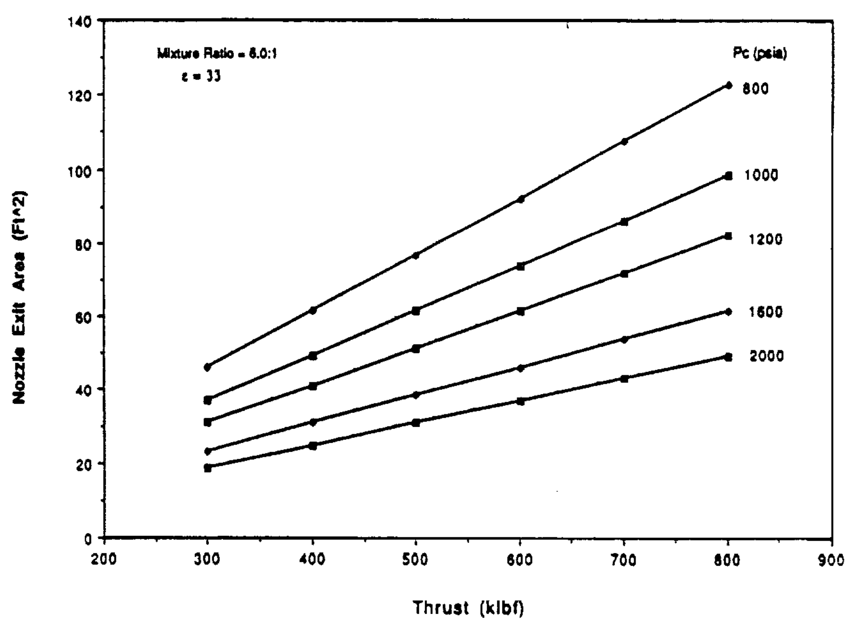


FIGURE 2.2-9 ENGINE LENGTH AND NOZZLE EXIT AREA AS A FUNCTION OF VACUUM THRUST AND CHAMBER PRESSURE

Reliable estimates of engine weights is probably the most difficult engine characteristic to parameterize in this effort, as a significant portion of the engine weight, the installation plumbing, is extremely vehicle dependent. Table 2.2-1 shows a comparison of the engine component weights extracted from an SSME weight table (dated 5-23-83) versus those generated by the ASCEM code. We have also listed the basis for each component weight estimate, denoting a direct calculation (CALC). An estimated scaled from the weight estimates of other components (SCALED), or a hybrid, wherein calculations are performed with some scaling required (CALC/SCALED). As shown in Table 2.2-1, the component weight estimates can be matched almost exactly for the SSME. The installation plumbing is based on installing the SSME in the STS Orbiter. Other SSME vehicle applications, or advanced/alternate engine designs for vehicles with different installation, functional, and performance requirements may result in installation plumbing weights that vary significantly from those for the SSME in the STS Orbiter.

As a result, engine weight estimates to be given in this report for both split expander and full flow staged-combustion engine cycles do not include installation plumbing, which must be estimated for the specific vehicle application under consideration.

TABLE 2.2-1 ENGINE DESIGN MODEL WEIGHT COMPARISON

(WEIGHTS ~ LB_M)

COMPONENTS	SSME (5/23/83)	ASCEM PREDICTION	BASIS
CHAMBER	466	465	CALC / SCALED
NOZZLE	1328	1332	CALC / SCALED
INJECTOR	394	395	CALC / SCALED
LOX TURBOPUMP	574	573	CALC
LOX BOOSTPUMP	199	195	CALC
LH_2 TURBOPUMP	775	773	CALC
LH_2 BOOSTPUMP	176	176	CALC / SCALED
PREBURNERS	229	229	SCALED
MAIN FEED VALVES	223	221	CALC
MISC VALVES / PLUMBING	809	809	SCALED
AVIONICS & CONTROLS	363	366	SCALED
SUBTOTAL	5536	5531	
INSTALLATION PLUMBING	1421	1421	SCALED
	6957	6952	

Total engine weight estimates versus chamber pressure and vacuum thrust are shown in Figure 2.2-10 for a mixture ratio of 6.0:1. Parametric data can also be generated for other mixture ratio conditions as will be shown in Section 2.3.

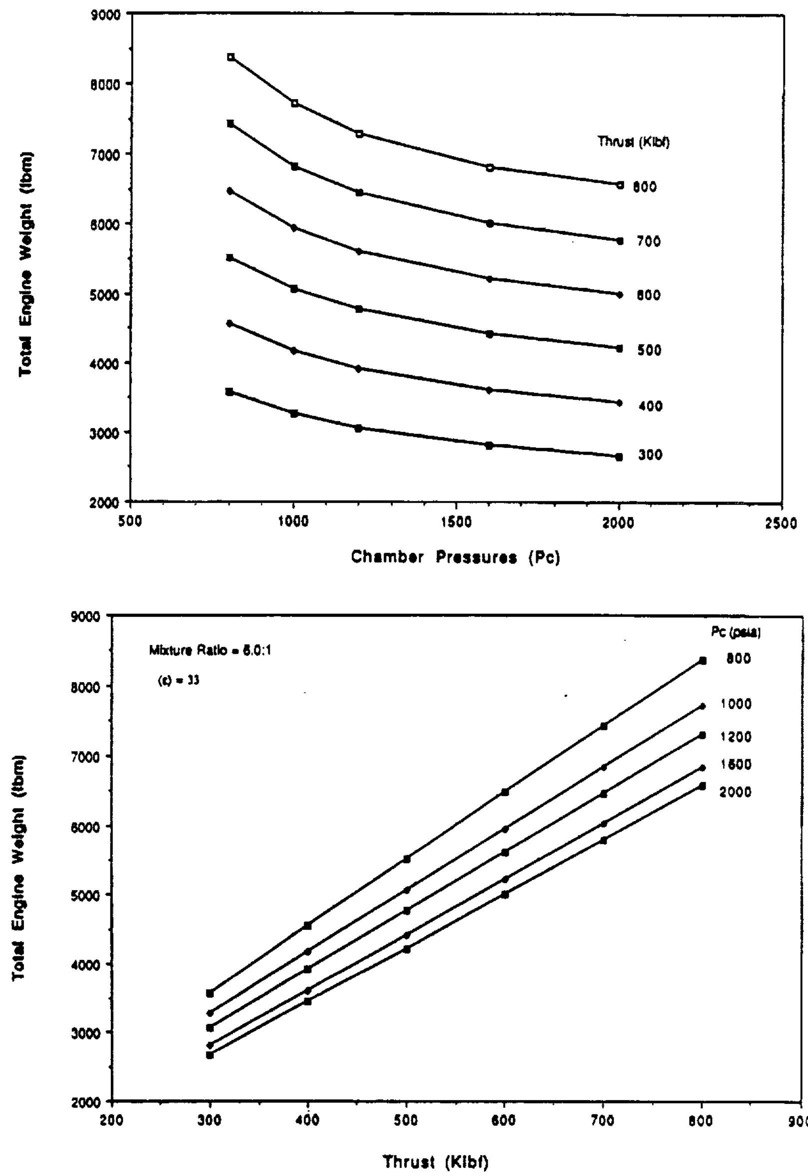


FIGURE 2.2-10 TOTAL ENGINE WEIGHT AS A FUNCTION OF CHAMBER PRESSURE VACUUM THRUST

2.2.2 Full Flow Staged Combustion Cycles

The Acurex Corp. developed parametric characteristics of the full flow staged combustion (FFSC) LOX-LH_2 rocket engine cycle. Parametric characteristics included such items as I_{SP} versus chamber pressure, size and weight versus thrust level and chamber pressure, and operating parameters such as turbine inlet temperature versus chamber pressure, etc. A hybrid option of the FFSC concept using staged combustion on the LH_2 side and expander operation of the LOX side, was also characterized.

The full flow staged-combustion cycle has only limited interaction between engine I_{SP} and the thermal and fluid efficiencies of the machinery. In this case the designer can choose to run the turbines as cool as possible, which largely eliminates material thermal limits as an issue. The dense slow moving gases can effectively drive low tip speed turbines, and produce only modest centrifugal stresses. The designer can focus on low cost, well-proven materials, derived from low cost, well proven processes, impose only modest (low cost) quality requirements, and manufacture to modest (low cost) tolerances.

The selection of the full flow combustion cycle (FFSC) could result in a low cost approach. The cost control forces can dominate decisions, because there is little compelling incentive for high cost program elements and a practical engine can be produced using low cost approaches.

2.2.2.1 Cycle Schematics

Adaptable to various propellant combinations, (LOX/LH_2 is shown in Figure 2.2-11); this rocket cycle uses fuel-rich combustion products to drive the full turbopump and oxidizer-rich combustion products to drive the oxidizer turbopump. All propellant flow is ultimately burned and exhausted through the main chamber throat giving this engine the maximum I_{SP} available. All propellant flow is used to drive the turbopumps, resulting in the lowest possible turbine inlet temperatures for a given main chamber pressure. Design studies of engines based on this cycle show such engines to be of noteworthy simplicity, with the potential of offering low cost without compromising reliability, safety, or performance.

A hybrid version of the FFSC cycle, which eliminates the oxidizer-rich preburner, and substitutes oxidizer heated in a regenerative cooling jacket of the main chamber to drive the oxidizer turbopumps, while retaining a fuel-rich preburner to power the fuel turbopump is shown schematically in Figure 2.2-12.

This cycle may be more simple in some respects than the FFSC cycle because it only has a single, fuel-rich, preburner. However, it is limited in chamber pressure to on the order of 2000 psig by the cooling capability of the oxidizer, and is heat transfer limited on engine size, to less than 450K. It is also not well suited to high or variable mixture ratio operation, because of the cooling function limitations.

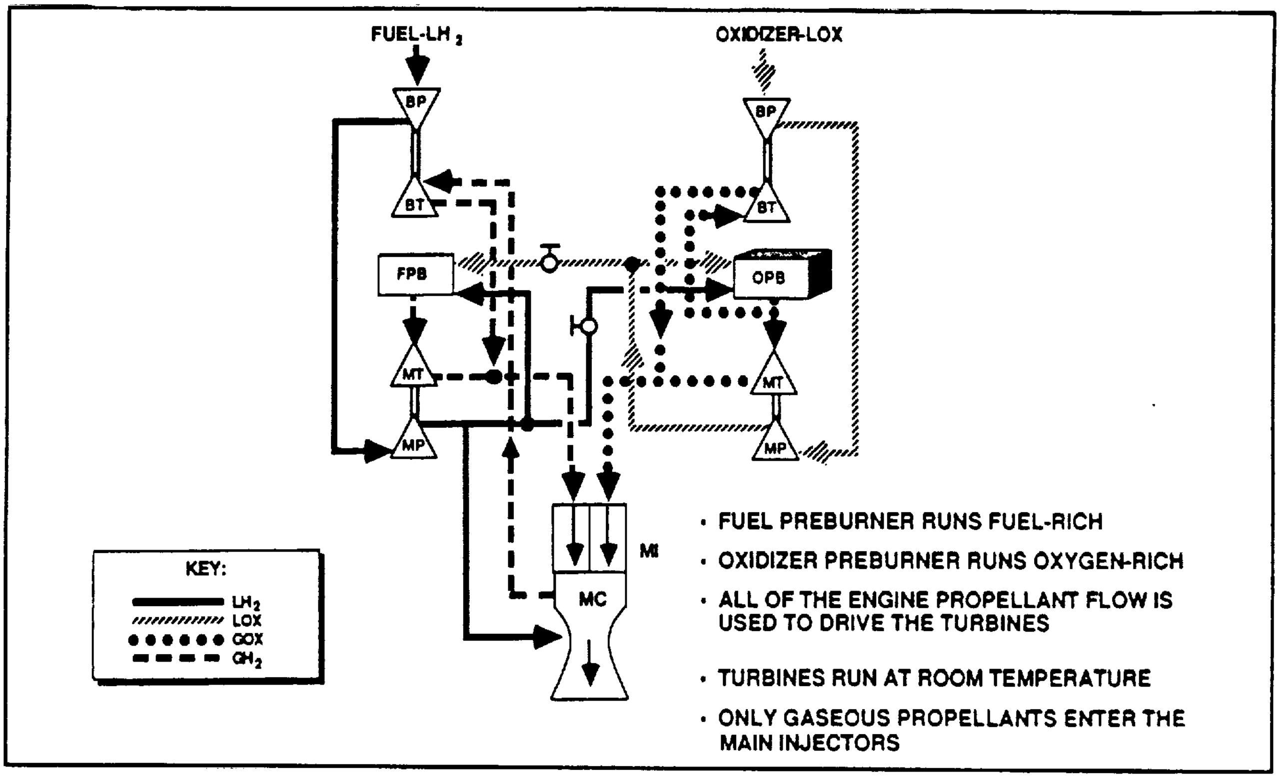


FIGURE 2.2-11 FULL FLOW STAGED-COMBUSTION CYCLE SCHEMATIC

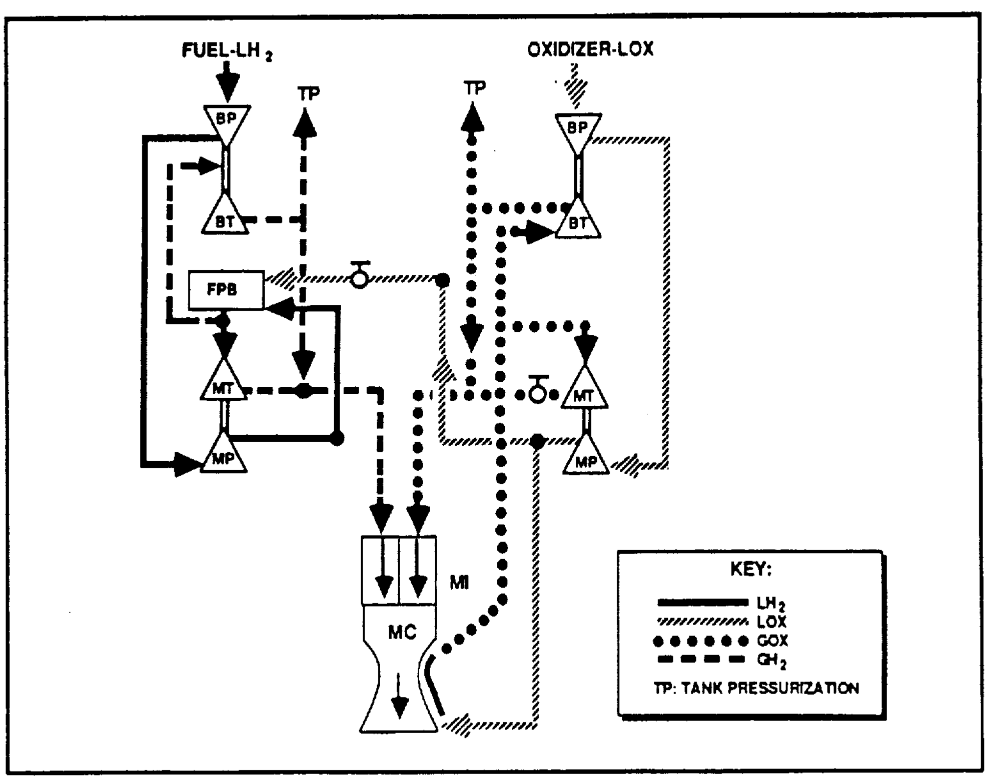


FIGURE 2.2-12 FULL FLOW/EXPANDER HYBRID CYCLE SCHEMATIC

2.2.2.2 Parametric Data

The full flow cycles have the potential to offer higher ISP for a given chamber pressure, mixture ratio and expander area ratio, than open LOX/LH_2 cycles such as the gas generator cycle. This is because overboard bleed losses are avoided, gas-gas main injection offers high combustion efficiency, combustion stability can be high as a result of the gaseous injection so the acoustic liner with its losses can be eliminated, and uniform, rapid, localized combustion will minimize film cooling losses. Typical ISP versus mixture ratio is shown in Figure 2.2-13. The effect of chamber pressure on ISP is shown in Figure 2.2-14. These figures are applicable to both the FFSC and hybrid full flow/expander option.

The high mass flows available to drive the turbines results in low turbine inlet temperatures. These low temperatures keep volumetric flow moderate so turbine sizes remain practical even at the lower chamber pressures. Figures 2.2-15 shows turbine inlet temperature as a function of chamber pressures for the FFSC cycle. Figure 2.2-16 shows the range of turbine temperatures for both LOX and LH_2 turbopump turbines for the hybrid cycle variant. Note there is an upper limit to the range for this cycle based upon practical values of thrust chamber heat transfer/cooling.

Engine size parameters are shown on Figure 2.2-17. Two interesting findings are that for a fixed L*, the length of a family of combustion chambers remains constant over a range of chamber pressures and thrust levels. This comes about because there is a fixed relationship between injector face area, which sets chamber diameter, and throat area for a given thrust and Pc, which then establishes the chamber length for a given L*.

Another finding was that the diameter of the powerhead, including laterally mounted turbopumps, remained constant over the range of chamber pressure for a given level. As chamber pressure increased, chamber diameter decreased, but the decrease in chamber diameter was closely offset by increased turbopump sizes so overall dimension change was negligible.

Engine weight parameters were derived from a family of FFSC designs which were configured in sizes from 650K to 1500K lbs thrust. Weights for this family of designs were estimated by taking dimensional data form the drawings. It is believed to be accurate within approximately ±5%. Figure 2.2-18 shows the correlation of these weights with other known engines. This correlation appears to confirm the weight estimates.

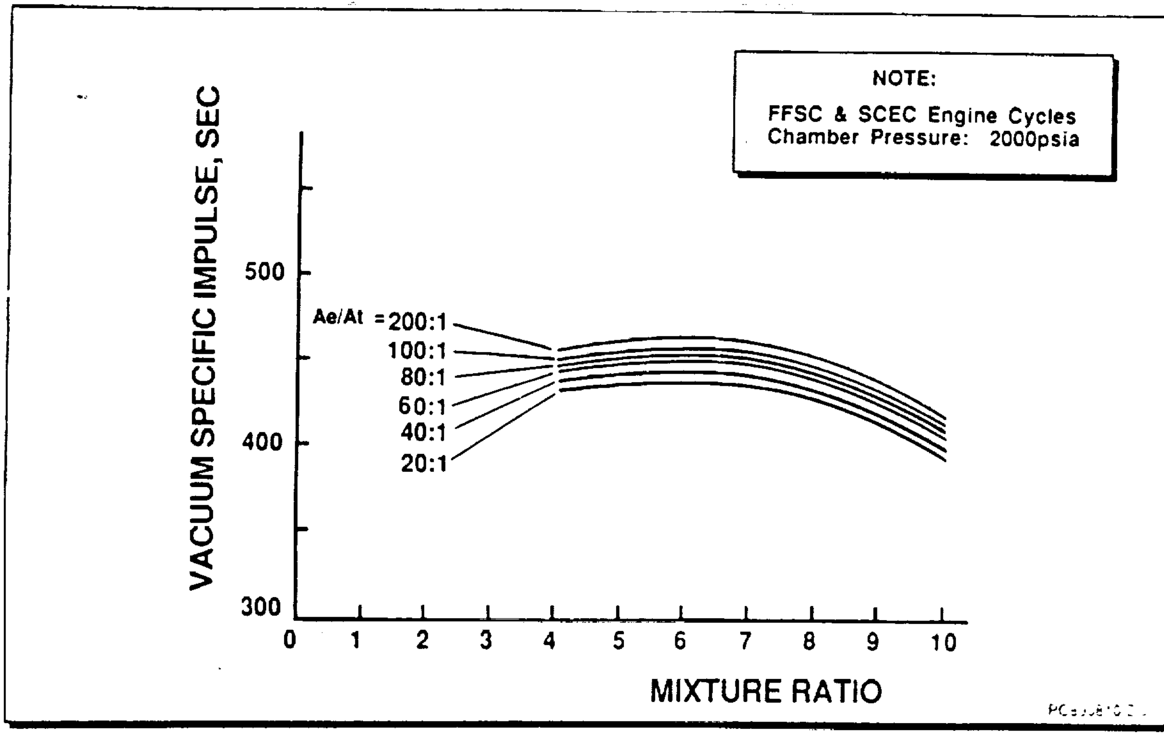


FIGURE 2.2-13 VACUUM SPECIFIC IMPULSE VERSUS MIXTURE RATION AND NOZZLE EXPANSION RATIO

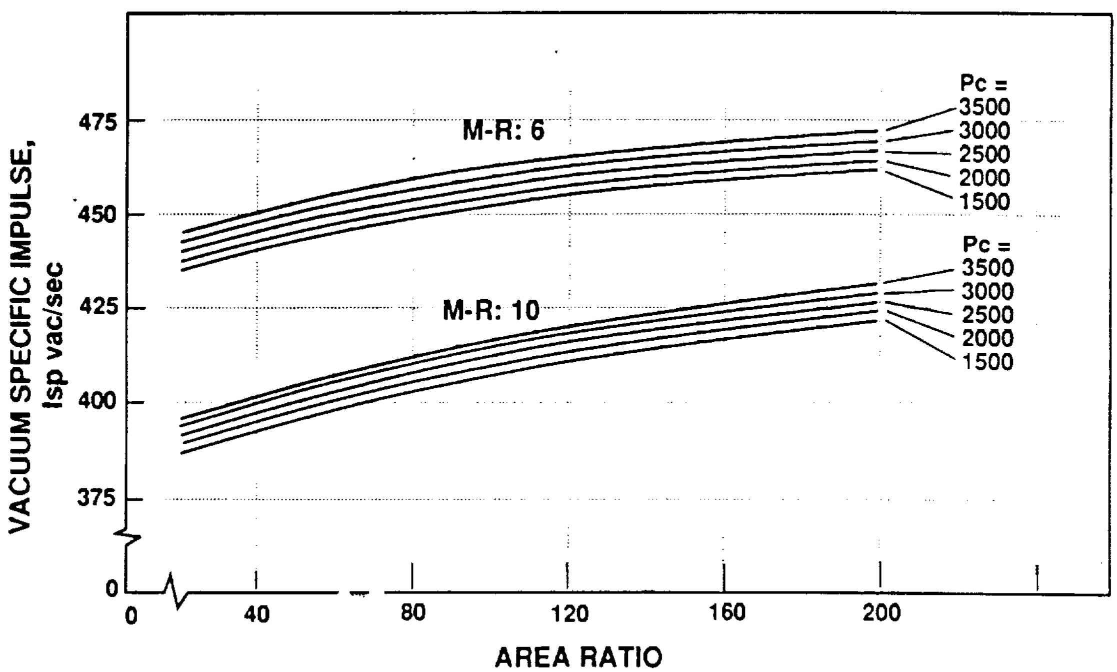


FIGURE 2.2-14 CHAMBER PRESSURE EFFECTS ON VACUUM SPECIFIC IMPULSE

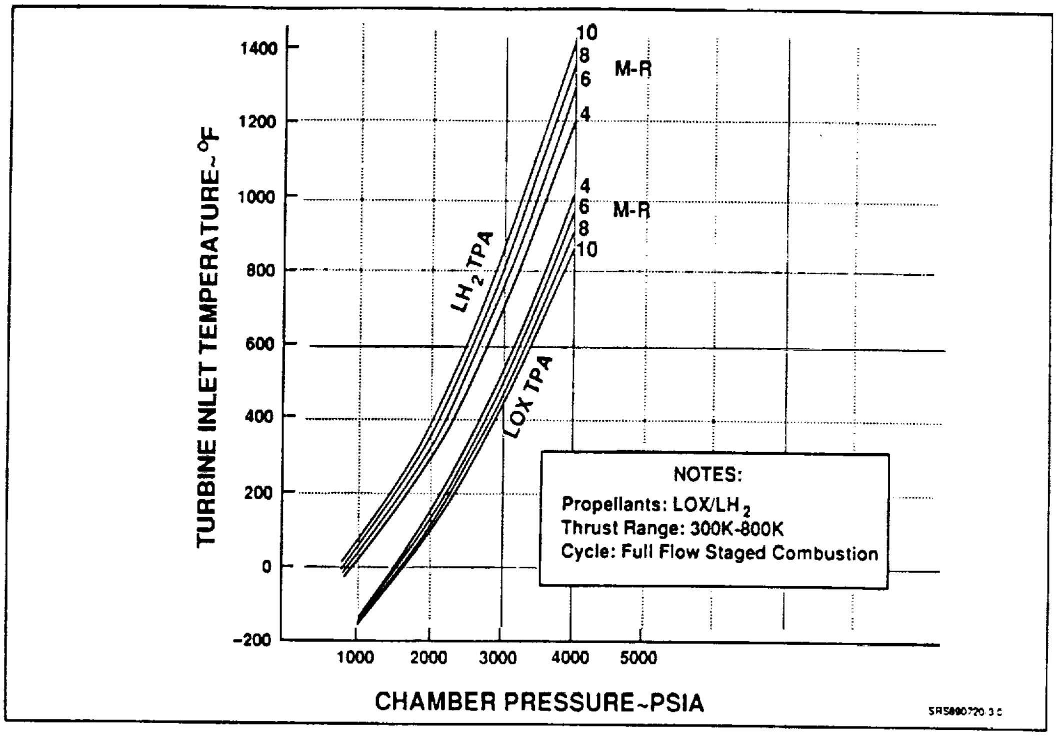


FIGURE 2.2-15 FFSC TURBINE INLET TEMPERATURE VERSUS CHAMBER PRESSURE

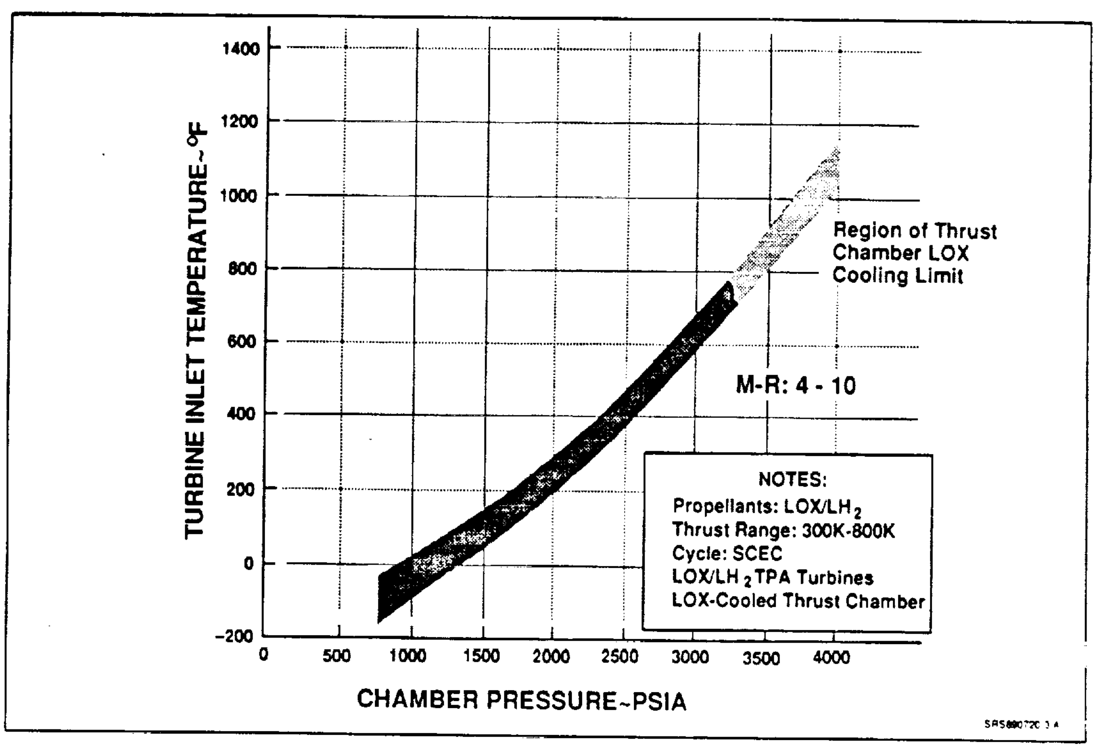


FIGURE 2.2-16 HYBRID FULL FLOW/EXPANDER OPTION TURBINE INLET TEMPERATURE VERSUS CHAMBER PRESSURE

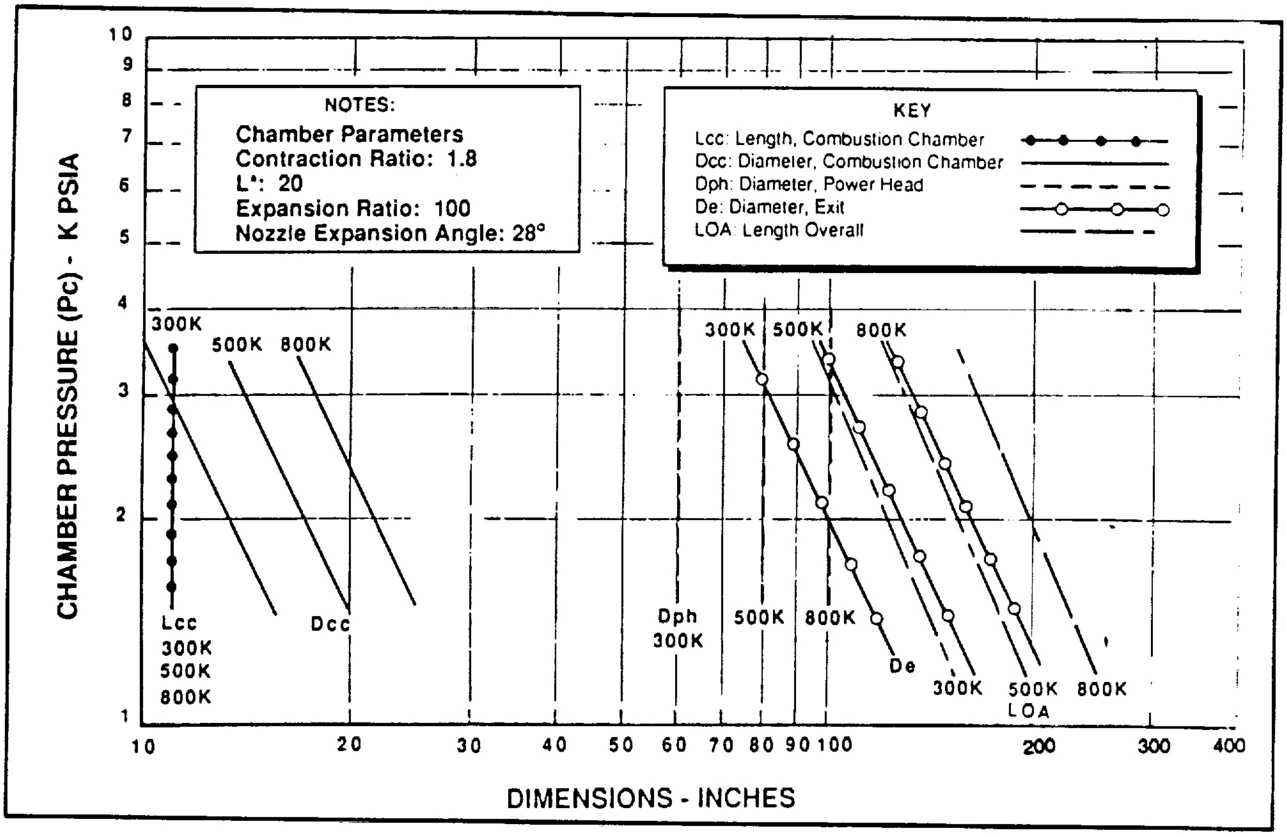


FIGURE 2.2-17 ENGINE SIZE PARAMETERS

Figure 2.2-19 shows the effect of chamber pressure on component weights. This chart is useful in that it can be applied over a wide thrust range by virtue of the weight correction factor versus thrust curve provided. Figure 2.2-20 combines the data of Figure 2.2-19, to give a total engine weight versus thrust for a range of chamber pressures.

The hybrid full flow/expander option would weigh about 96% of the FFSC cycle, because of the elimination of the LOX turbopump preburner.

Point design FFSC engines at 580K (STME size) and 470K (SSME size) were sized as shown in Figure 2.2-21. Envelope dimensions grew from the 470K thrust level to the 580K thrust level.

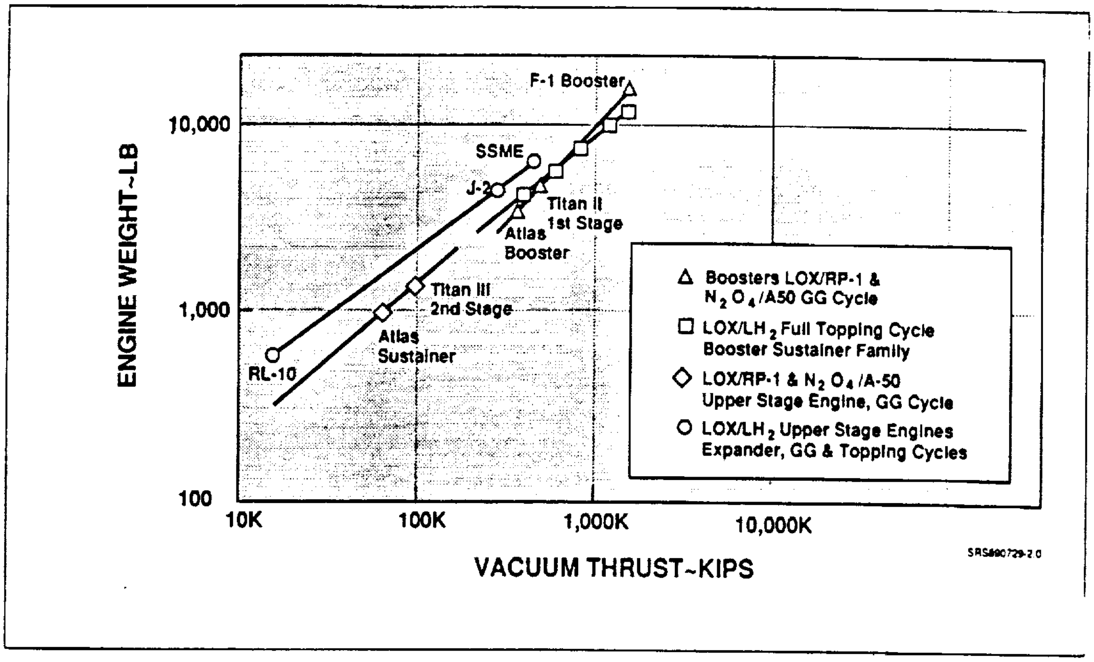
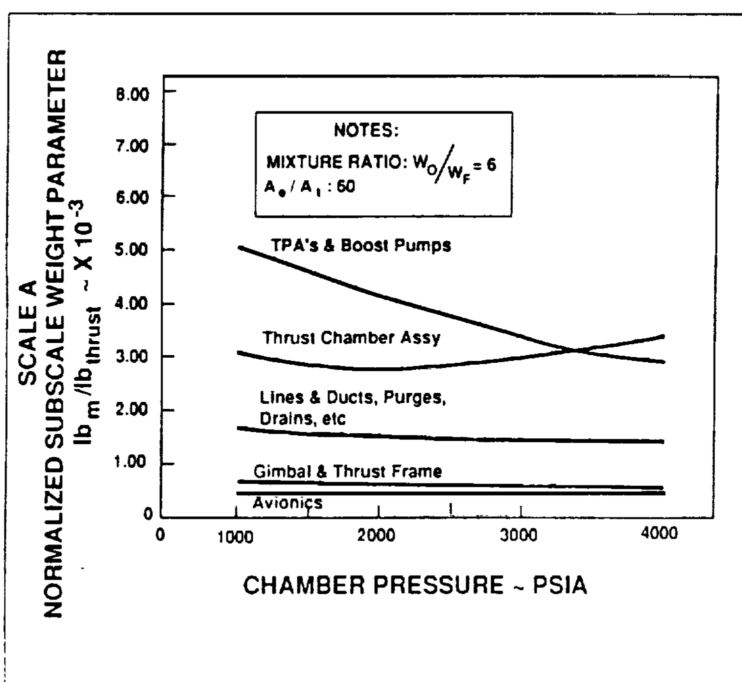


FIGURE 2.2-18 HISTORICAL ENGINE THRUST-TO-WEIGHT DATA



NOTE:
SUBSYSTEM WEIGHT CALCULATION

Subsystem Weight W_{SS} :
W_{SS} = Thrust X Scale A X Scale B

Where Scale A is read for a given Pc and Scale B is read for a given Thrust

FIGURE 2.2-19 FFSC PARAMETRIC SUBSYSTEM WEIGHTS

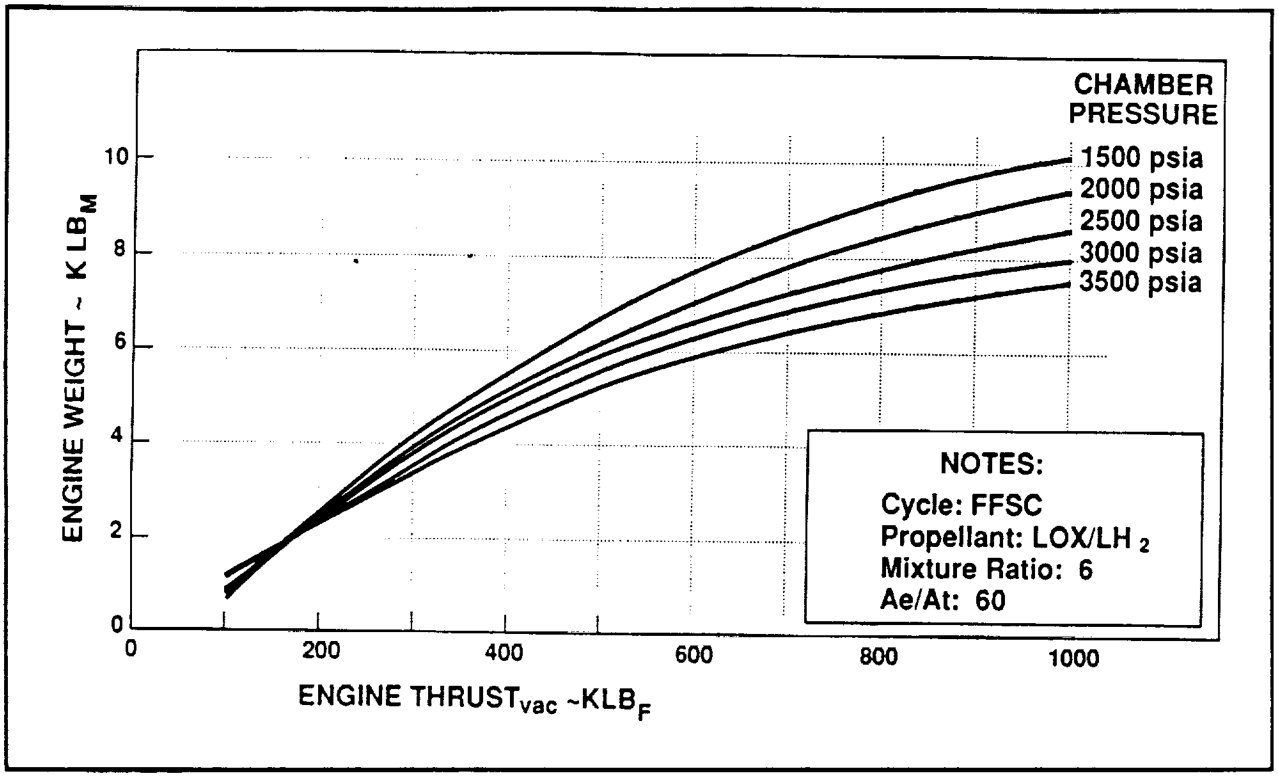


FIGURE 2.2-20 FFSC TOTAL ENGINE WEIGHT VERSUS VACUUM THRUST

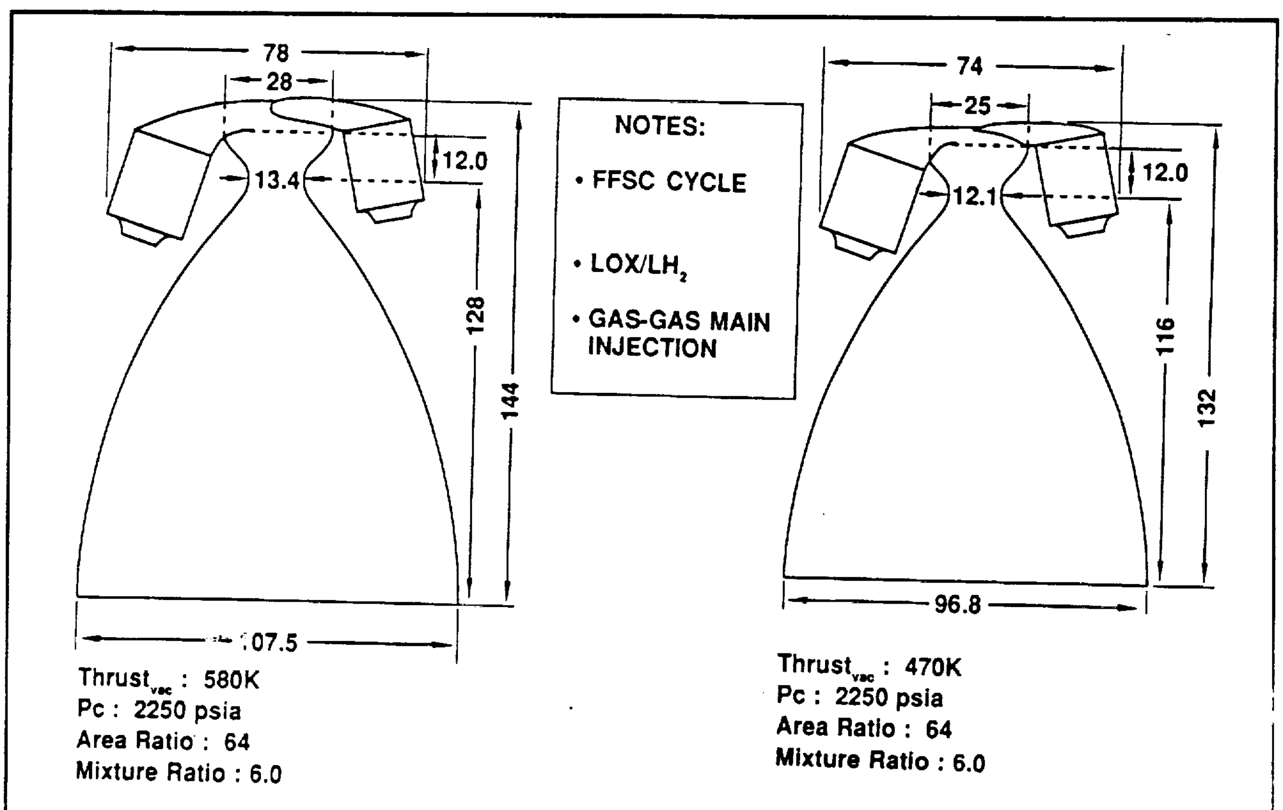


FIGURE 2.2-21 POINT DESIGN ENGINES

Table 2.2-2 shows the engine design point information for the two engine thrust levels, as well as propellant feed system parameters for each (Table 2.2-3).

TABLE 2.2-2 FFSC DESIGN POINT PARAMETERS

PARAMETER		THRUST LEVEL 470K	580K
Mixture Ratio (Ox/Fuel)		6.0	6.0
Isp vac	(sec)	453	453
$\dot{W}$ Total	(lb_m/sec)	1033	1275
$\dot{W}$ Fuel (LH_2)	(lb_m/sec)	147.6	182.1
$\dot{W}$ Oxid (LOX)	(lb_m/sec)	885.4	1092.9
Pc	(lb_f/in^2)	2250	2250
Nozzle Expansion Ratio	(ε)	64	64
$C_{f\,vac}$ (γ=1.25)		1.85	1.85
A_t	(in^2)	112.9	139.3
D_t	(in)	11.99	13.32
D_e	(in)	95.92	106.56

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TABLE 2.2-3 FFSC PROPELLANT FEED SYSTEM PARAMETERS

PARAMETER		THRUST LEVEL 470K Oxid	Fuel	580K Oxid	Fuel
ΔP inj (@ 10% stiffness)	(lb_f/in^2)	225	200	225	200
ΔP Lines and Values	(lb_f/in^2)	200	100	225	200
P_{te}	(lb_f/in^2)	2675	2575	2675	2575
Turbine Pressure Ratio		1.60	1.50	1.60	1.50
P_{ti}	(lb_f/in^2)	4280	3860	4280	3860
Preburner ΔP	(lb_f/in^2)	100	60	100	60
P-B Injector ΔP	(lb_f/in^2)	400	350	400	350
P-B Inj. Inlet	(lb_f/in^2)	4780	4270	4780	4270
Line drop ΔP	(lb_f/in^2)	80	30	80	40
Pump Exit Pressure	(lb_f/in^2)	4860	4300	4860	4300
Pump Inlet Pressure	(lb_f/in^2)	60	40	60	40
Pump ΔP*	(lb_f/in^2)	4800	4260**	4800	4260**
Pump Efficiency	(%)	78	74	80	76
Pump Power	(SHP)	20,015	50,560	24,090	60,740
$\dot{W}_{turb}$	(lb_m/sec)	808	180	996	222
Turbine Pressure Ratio		1.60	1.50	1.60	1.50
P-B MR (Ox/Fuel)		120	0.78	120	0.78
γ (Ratio of Specific Heats)		1.35	1.36	1.35	1.50
C_p		.25	3.00	.25	3.00
Turbine Efficiency	(%)	70	68	70	68
T_{ti}	(°R)	840	996	736	970

* Boost pump and main pump lumped together.

** Note: LH_2 chamber coolant flow is in parallel, not series, so is not an " added" pressure drop.

2.3 Assessment Versus Alternate Concepts (Subtask 3)

In this task the split expander and full flow staged-combustion cycles were assessed relative to both subsystem technologies and vehicle system applications.

2.3.1 Subsystem Technology Assessment

The state-of-the-art of the subsystems of both the split expander cycle and the full flow staged-combustion cycle are assessed in the following subsections.

2.3.1.1 Split Expander Cycle

Several of the technology issues related to the split expander cycle were previously listed in the summary of characteristics and issues in Section 2.2.1. The technology issues that will be discussed in this section include: ongoing and forthcoming programs, split fuel flow (unproven concept), startup on tank head (no boost pumps) and restart, test program, manufacturing, safety and reliability, materials, and other performance parameters.

The ALS program has identified key technology, which development concerns include:

- Available power, which is driven by heat transfer/pressure drop, may require enhancements/coatings, (i.e., the power margin or the heat transfer to the coolant for turbine drive versus the chamber wall temperature and chamber pressure requirements).
- Combustion stability margin at low pressure for large combustion chambers. [This has been successfully accomplished in several other (non-split expander) rocket engine development programs, i.e., the F-1 engine - 1,522,000 lbs thrust @ 982 psia, the MA-5 booster - 377,500 lbs thrust @ 639 psia, and the J-2 - 230,000 lbs thrust @ 763 psia, the RS-27 - 207,000 lbs thrust @ 702 psia, and the H-1 - 205,000 lbs thrust @ 700 psia.]
- Start repeatability and stability as the chamber goes through phase changes (start transient characteristics).

The effect on thrust as a function of chamber pressure for different combustion chamber materials is shown in Figure 2.3-1. This figure shows that a copper chamber can operate at a higher Pc at a given thrust level than either Haynes or stainless steel. This is explained by:

Q/A is the convective transfer rate per unit area

$$Q/A = h_g(T_{HG} - T_{HW}) = k/t(T_{HW} - T_{CW}) = h_c(T_{CW} - T_C)$$

Q = Heat

A = Area

T_{HG} = Temperature of hot gas

T_{HW} = Temperature at hot wall

T_{CW} = Temperature at cold wall

T_C = Temperature of coolant

h_g = Hot gas film coefficient

h_c = Film coefficient for liquid side film

k = Thermal conductivity of the wall

t = Wall thickness

where Q/A is set by the hot gas conditions. The allowable T_{CW} is set by tube thickness and conductivity and maximum allowable T_{HW}. Coolant pressure loss is set by required T_{CW}.

Figure 2.3-1 shows that for maximum chamber pressure versus vacuum thrust, maximum chamber pressure (MPC) must decline as the thrust increases. However, this levels out near the 800 Klb region. The ranges are from ~1575 psia @ 100 Klb to ~1250 psia @ 800 Klb.

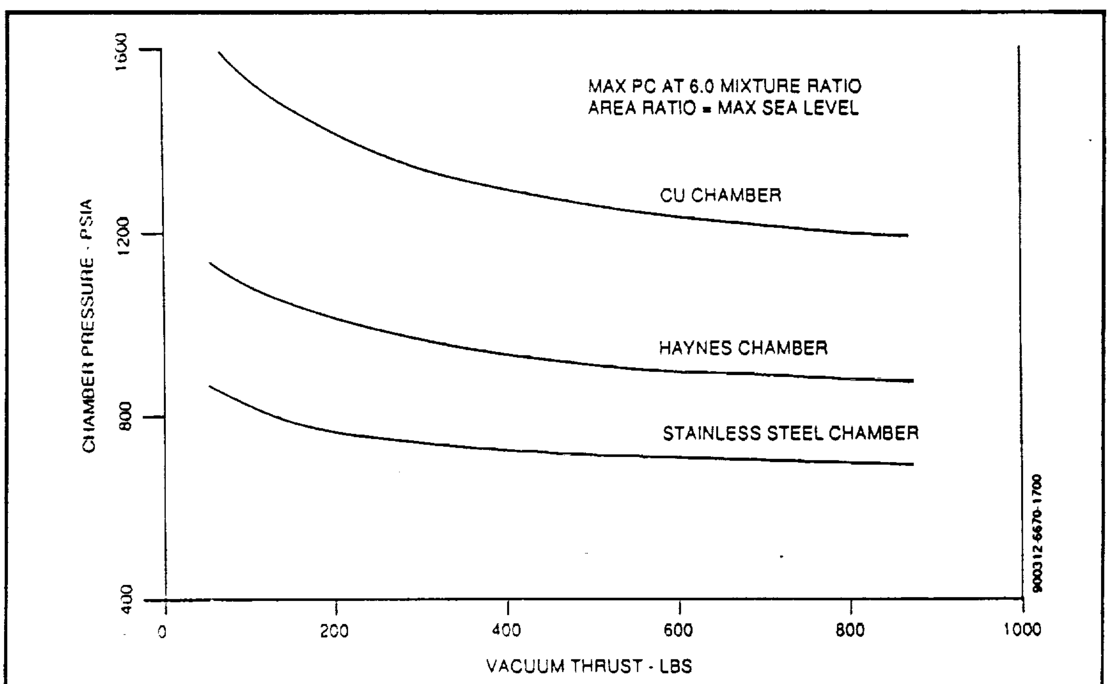

FIGURE 2.3-1 CHAMBER PRESSURE CAPABILITIES OF CANDIDATE CHAMBER MATERIALS VERSUS THRUST

The low engine chamber pressure allow for simpler LH_2 and LOX pumps made with non-exotic materials. Some system peculiar parts will be required, but the technology base is already available. The hydrogen pump is a two stage pump, and at full power, the majority of the flow passes through the first stage then directly to the chamber. The other part of the fuel flow passes through a second stage before picking up heat to drive the turbines. The LOX pump is a one stage pump. Since the split expander is a self limiting cycle, the turbine is limited by heat transfer in the regeneratively cooled nozzle. The split portion of the fuel flow that passes through the regeneratively cooled nozzle drives both turbines (fuel then LOX) before finally being mixed with the remainder of the flow. The lower chamber pressures place no significant technology development requirements on the turbine.

The combustion stability margin is an area of concern where some technology development will be required. The relatively low chamber pressure combined with a large chamber will require analysis and testing to ensure stability margins. However, as previously mentioned other engines with large chambers and low chamber pressures have been successfully developed. Start repeatability and stability as chamber goes through phase change (start transient characteristics) may also present a need for technology development.

Several chamber and nozzle wall materials are being considered. Materials with high heat transfer capability are desirable to minimize pressure drop. However, materials with higher heat transfer capability tend to be heavier and more costly. Technology development in this area will be needed to provide lightweight, low cost engines. The cost and weight of the chamber is also a function of the chamber pressure and the type of chamber material. Figure 2.3-2 shows the stainless steel chamber is the least costly and weighs the least, the Haynes tubular chamber is heavier and slightly more costly, while the copper tube chamber is the heaviest and the most costly.

System peculiar line, valve, ignitor, etc. components will have to be designed and built, but no significant technology development impacts are expected.

2.3.1.2 Full Flow Staged-Combustion Cycle

The full flow cycle can provide benefits within near-term time frames and have a high level of predictable success and at modest cost. The full flow concept has the potential to be a highly reliable generic propulsion system that can be built in various sizes and cluster arrangements to serve a wide range of NMTS applications. The following is an assessment of the state-of-the-art of the subsystem technologies of the full flow cycle.

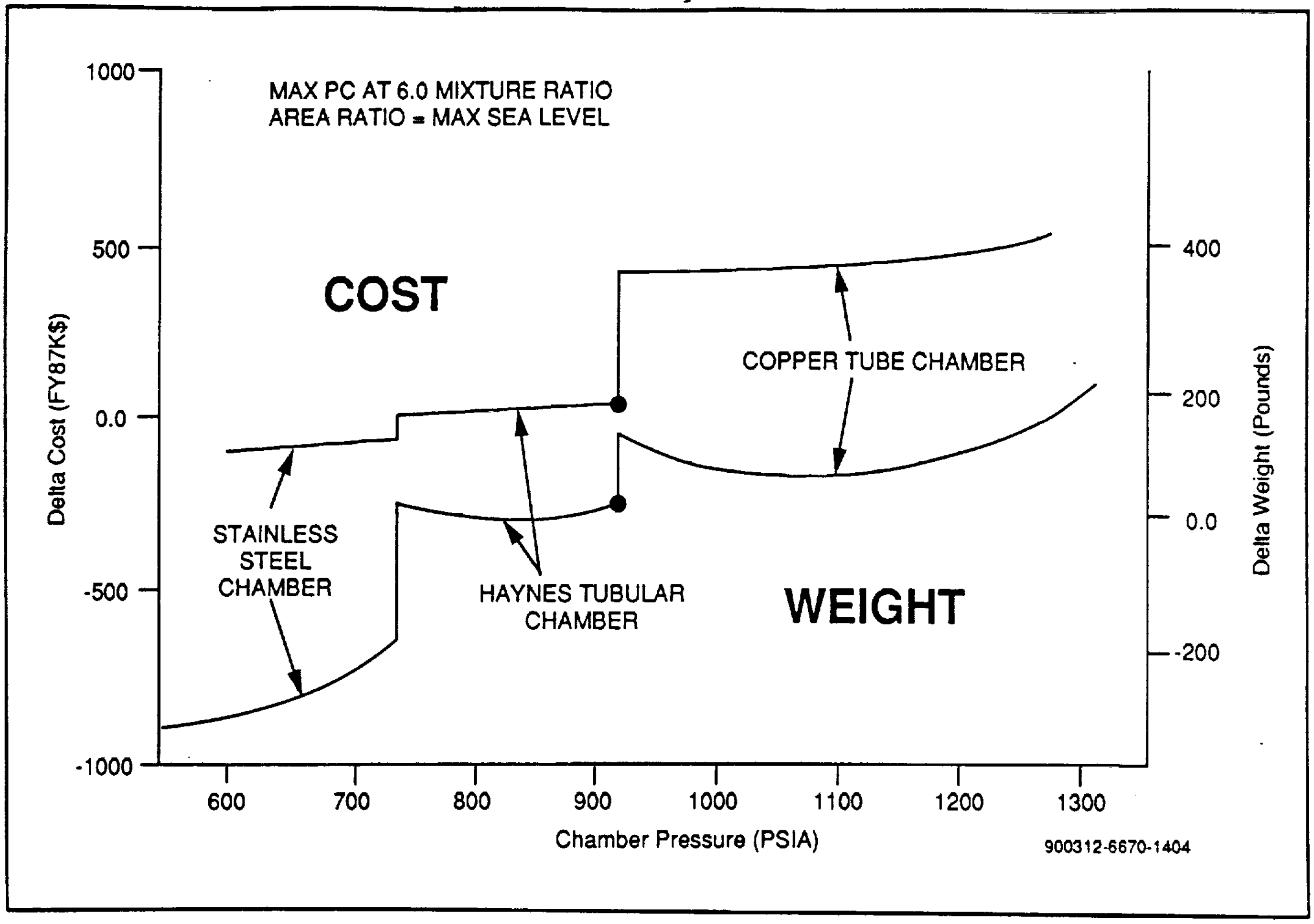

FIGURE 2.3-2 CHAMBER MATERIALS AND PRESSURE EFFECTS ON COST AND WEIGHT

2.3.1.2.1 Turbomachinery

The very high mass flow through the turbine and low turbine inlet temperature allows the turbine tip speed and shaft speed to be lower than high temperature designs. This reduces centrifugal stresses and bearing DN values. On the hydrogen turbopump, the turbine is cool enough to permit the use of material such as A286, which are not subject to hydrogen embrittlement.

On the LOX turbopump side, the cool dense gas permits the use of a simple single stage impulse turbine. A single stage, single inlet pump impeller can be used even though shaft speeds are moderate. The LOX-cooled bearing DN is less than 1×10^6, a very conservative value. A simple lift-off seal serves to prevent shaft leakage during chill-down. A conventional labyrinth serves to keep the operating pressure in the bearing cavity above critical pressure during operation. This avoids two-phase flow conditions within the bearings and the attendant loss of cooling capacity. Overall, this cycle leads to simple reliable low cost turbomachinery.

Current state-of-the-art is exemplified by the SSME turbopumps. These LOX and hydrogen turbopumps have been under development for nearly two decades and have received the attention and evaluation of a wide spectrum of the aerospace community. These turbopumps operate at temperatures, stress levels, bearing DN's, suction specific speeds, and thermal transients all well above those which occur in the full flow staged combustion engine. Materials requirements for hydrogen embrittlement prevention place special demands on the turbines. These requirements are not present in the full flow concept. The SSME autogenous LOX heat exchanger poses materials oxygen compatibility issues of the same magnitude as those that are present in the full flow cycle. In summary, the technology requirements for the turbomachinery of the full flow concept falls well within current state-of-the-art.

2.3.1.2.2 Ducting

The LOX/LH_2 propellants are routed through the turbopumps as cryogenics only up to the preburners. Thereafter, the principal propellant flows are at temperatures of about only several hundred degrees F until they are ignited in the main combustion chamber. The direct benefit from these features is that the main propellant ducting to the combustion chamber can be simple, uncooled, reliable, and low cost.

Current state-of-the-art for the SSME lines includes high temperature ducting with liners and various cooling provisions. Cryogenic ducting with bellows-type flex-sections including internal and/or external bellows restraints, vacuum jacketing and other insulation methods are also on the SSME. The proposed ducting for a full flow cycle engine would be well within the current state-of-the-art, potentially requiring simpler, lower cost component and materials than now used.

2.3.1.2.3 Preburners

The preburners for the full flow cycle use a configuration wherein the actual combustion reaction is entirely surrounded by an in-flow of raw propellant, i.e., combustion is submerged in the major propellant flow as illustrated in Figure 2.3-3. The purpose is to enable combustion to proceed to completion, yet ensure that preburner wall temperatures remain cool.

Downstream mixing of combustion products and the diluent propellant serve to provide the mixed mean temperature desired for turbine drive.

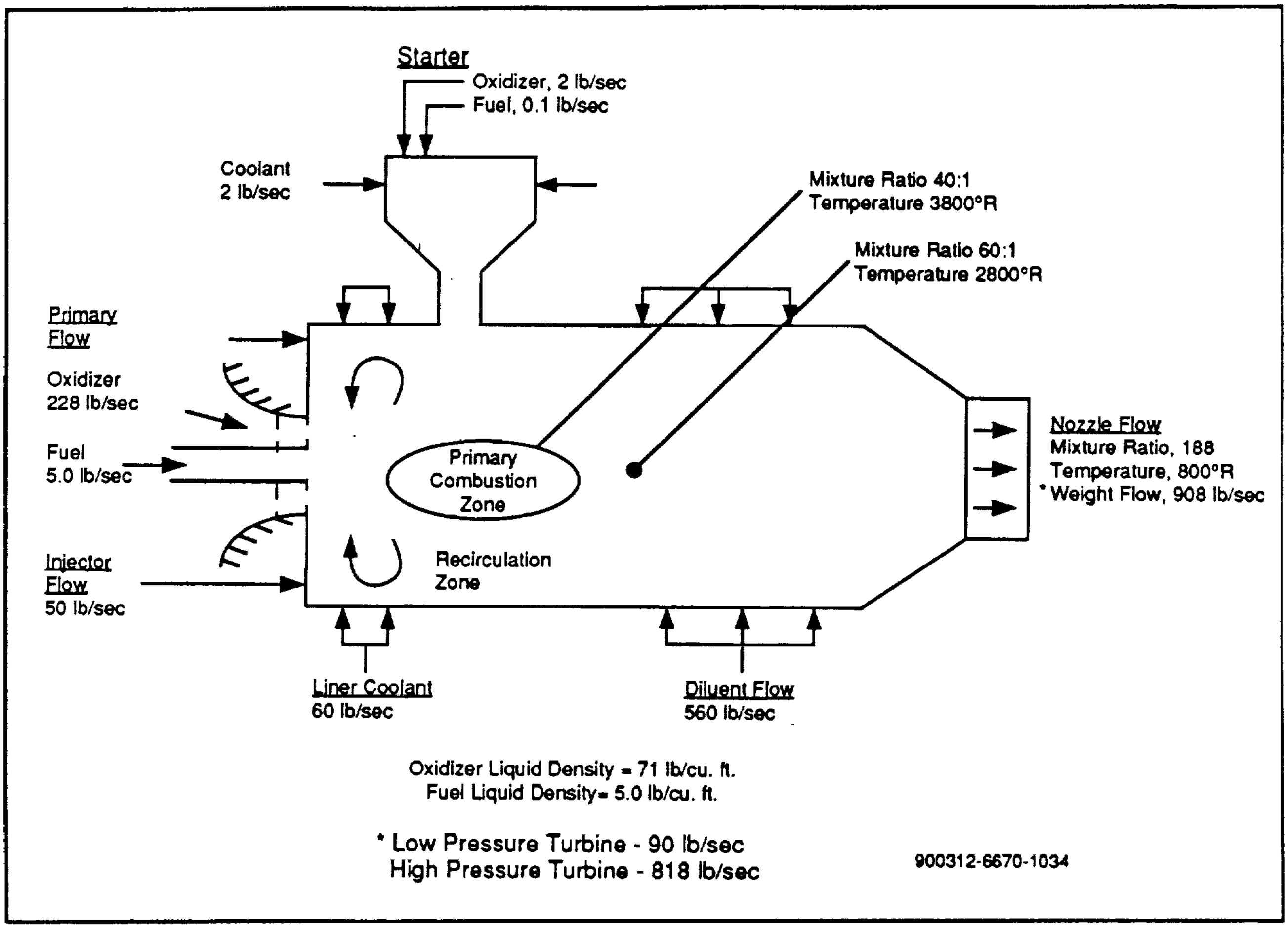

FIGURE 2.3-3 OXIDIZER-RICH PREBURNER

Both highly fuel-rich and highly oxidizer-rich combustion of LOX/LH2 have been demonstrated on successful NASA programs. Current state-of-the-art is to inject both propellants locally at the final mixture ratio and allow mixing and burning to occur simultaneously. Problems occur with uniform temperature control and in obtaining complete combustion. However, hydrogen-rich preburners drive the SSME turbines, and the MSFC oxidizer-rich experience was reported favorably. Figure 2.3-4 shows the wide range of mixture rations successfully fired at 1000 psia chamber pressure by MSFC in 1965. This provides a basis to undertake further high pressure high mixture ratio testing, with oxygen-hydrogen.

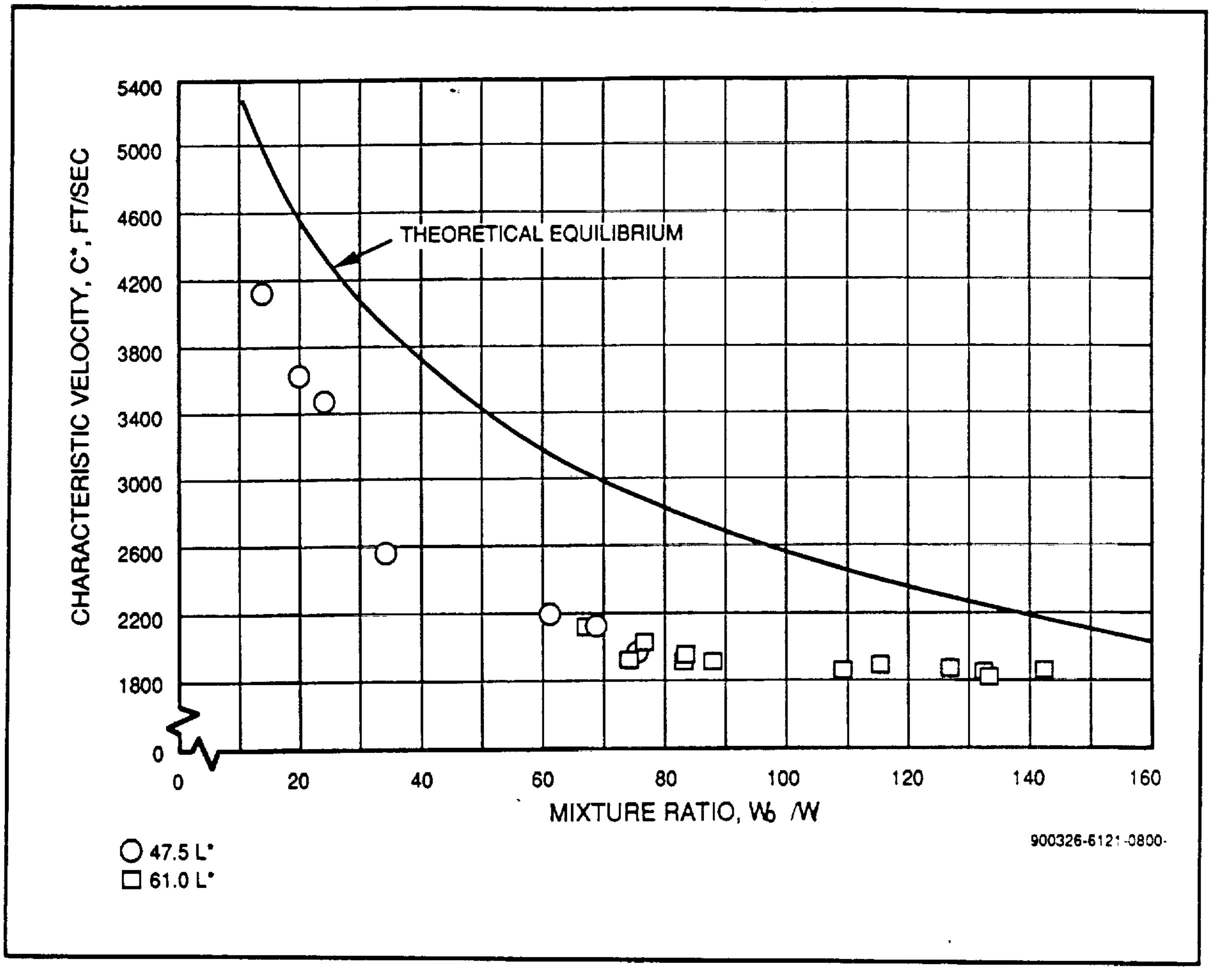

FIGURE 2.3-4 MSFC HIGH MIXTURE RATIO TEST DATA

2.3.1.2.4 Main Injector

In the full flow topping engine cycle at 2000 psig chamber pressure, both propellants arrive at the main injector at moderate (100-200°F) temperatures and as compressible fluids at approximately equal pressures. As a result, the main injector does not have severe thermal or pressure gradients. While yet to be demonstrated, it is envisioned that the main injector can be configured as simple divider plates since the more conventional spray injection required to mix liquid propellants is not needed. Rather, gas-phase mixing and combustion will be involved.

Current state-of-the-art compared to the type of gas-gas injectors envisioned is illustrated in Figure 2.3-5. It is apparent that gas-gas injectors may be far less complex than current injectors. There is a history of experience with gaseous injection for small scale testing as a convenience to avoid the complexity of cryogenic liquid handling.

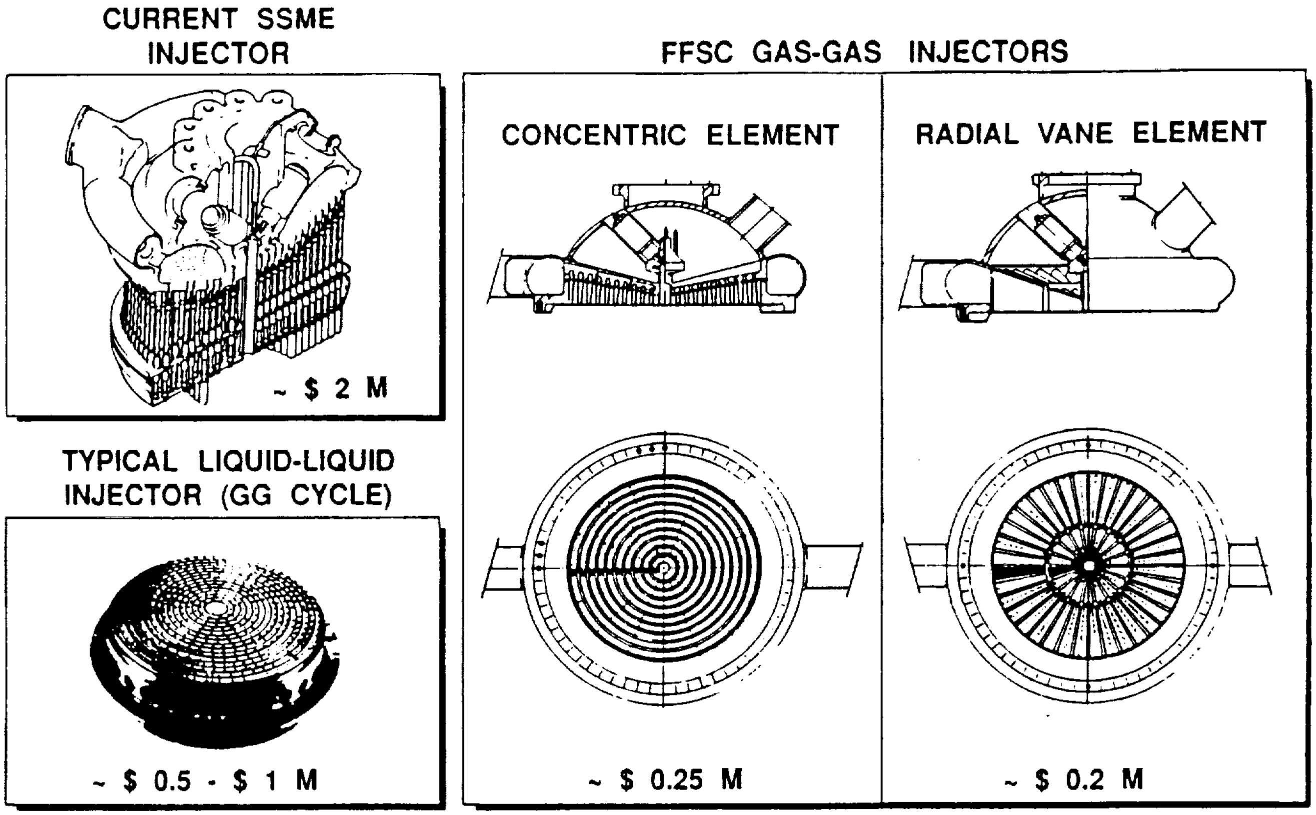

FIGURE 2.3-5 MAIN INJECTOR COMPARISON

2.3.1.2.5 Main Combustor

In the full flow staged combustion cycle both propellants are injected into the main chamber as partially burned, combustion ready gases. This avoids the ignition delay required for propellant heating and vaporization, and thereby avoids some of the phenomena that contribute to combustion instability. Furthermore, the fluid volumetric change in the gas-gas combustion process is only about one-tenth the change in volume occurring in going from liquid injectants to combustion products. And finally, since there is no liquid phase in the gas-gas combustion chamber, the instantaneous total mass of propellant in the chamber will be only about 50-60 percent of the equivalent liquid injected chamber which reduces the conditions capable of driving combustion instability modes. It is therefore anticipated that gas-gas combustion will be stable, rapid and uniform, with a well-defined combustion zone.

Since recirculation of combustion products is not required for propellant heating, a more uniform flow distribution in the chamber can be expected. This should avoid streaking as well as permit more effective film cooling since random turbulence in the chamber can be minimized.

2.3.1.2.6 Overall Design Philosophy

The full flow topping cycle performance, in terms of engine Isp, is not dependent on efficiencies of the turbomachinery components nor losses in ducts, valves and injectors. These losses are only reflected in turbine temperatures, and secondarily in engine weight, but not directly in mission performance.

It is possible to utilize this cycle characteristic to advantage by using less critical design features. Running clearances can be larger, manufacturing tolerances can be opened up, blading geometry can be less critical. Thrust balance means can be more effective and shaft/bearing systems can be simplified.

Because high mass flows are utilized in the turbine, turbine operating temperatures are low, and slower speed turbines can be used. Centrifugal stresses are reduced while material structural properties are increased.

Fracture mechanics and other structural considerations are minimized. Inspections for minute flaws in materials is not critical since larger flaws can be tolerated. Quality assurance requirements involving extreme precision in machining and in inspection steps can be relaxed to a more practical level, approaching industrial practices and costs.

Current state-of-the-art is well advanced in terms of all of the foregoing examples. The avoidance of high materials and manufacturing costs is one of the major advantages of this engine cycle. Analyses, and design detail, manufacturing and quality assurance precision, as well as test and operating margins are all less critical in this concept than conventional aerospace practice of today.

2.3.2 Vehicle Applications Assessment

SRS has performed limited analyses of engine applications in AMLS type vehicle concepts; this information is discussed in the following paragraphs. (Note: in the companion study "Propulsion Evolution Study" task, SRS has performed more extensive analyses of vehicle applications for gas-generator cycle (STME) and staged combustion cycle (SSME) engines, including applications in STS/Shuttle "C" and "Personnel Launch System or PLS" launch vehicles, as well as the AMLS concepts. Hopefully, further studies of these "undeveloped cycle engines" can include further examinations of vehicle applications).

As noted earlier, AMLS vehicle concepts are perceived as the next generation Space Shuttle type launch vehicle for the principal purpose of manned transportation to and from low Earth orbit. A range of vehicle concepts as used in NASA concept planning is shown in Figure 2.3-6.

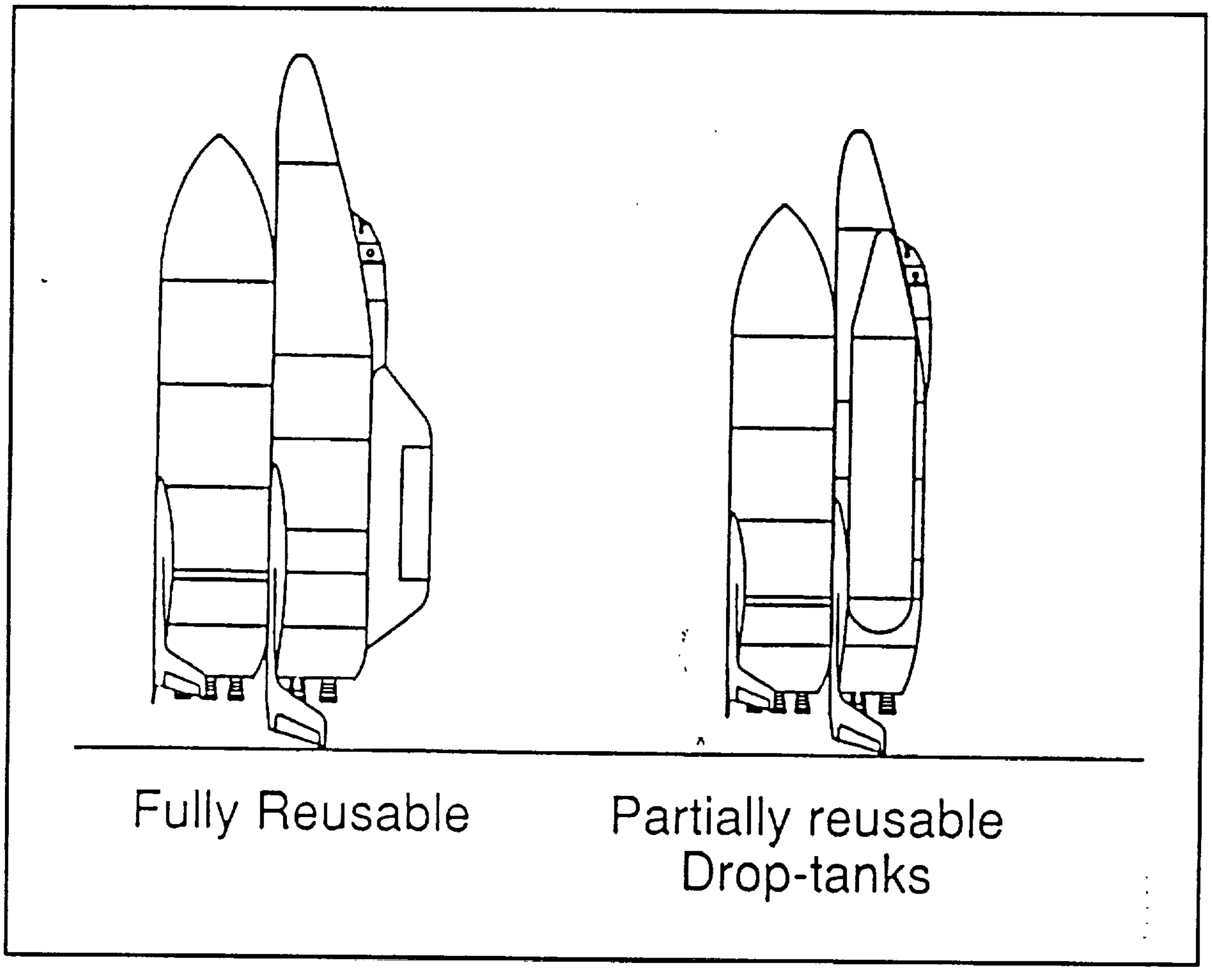

FIGURE 2.3-6 AMLS VEHICLE CONCEPTS

Varying degrees of vehicle recovery and reuse are being examined, with the two-stage fully reusable version being used currently as the baseline approach. Vehicle concept studies of AMLS vehicles have been relatively inactive during the time that this propulsion study has been in progress, with NASA vehicle concept studies concentrating during that period on Shuttle Evolution and PLS concepts and plans. A number of AMLS type vehicle concepts and sizes are available from previous studies, ranging from 20k to 65k payload capability, with some using hydrogen-fueled boosters and others using hydrocarbon-fueled boosters. We have used that data where possible, and have augmented with rough-order parametric sizing of AMLS type vehicles as a means to examine the effects of differing engine characteristics. We have used a nominal payload requirement in the Space Shuttle class, e.g., approximately 50k lbs to low Earth orbit for these analyses. There is considerable uncertainty at present in the inert weight characteristics of this class of vehicles, due both to the early state of design definition, and to uncertainties as to the levels of vehicle technologies that will ultimately be incorporated into these vehicles. In the companion "Propulsion Evolution Study", we have examined sensitivities to variations about the set of vehicle inert weight characteristics that we have used as "nominal"; in this discussion, we will use only those "nominal" values.

2.3.2.1 Engine Data and Characteristics

In this subtask, we have selected from analyses of three engine cycles a nominal set of characteristics for the purpose of future manned launch vehicle applications. Basic characteristics for two of these engine cycles (Split Expander and Full-Flow Staged Combustion) are shown in Figure 2.3.-7, in comparison with SSME engine data. During discussions of vehicle applications for these two engine concepts, we will indicate comparisons with vehicles using engines STME/STEP engines characterized for ALS vehicle applications, in addition to vehicles using SSME engines. (Note: we have continued in this study to use "STME" terminology to refer to hydrogen-oxygen engines designed for use in ALS launch vehicles). ALS and STEP (Space Transportation Engine Program) include a gas-generator cycle baseline version and two versions of split expander cycle (split expander reference engine and split expander alternate engine). As will be noted later, the split expander engine characterized in this study task for future manned vehicle applications will use combinations of chamber pressure and nozzle expansion ratio that are higher than those characterized for the less performance-sensitive ALS launch vehicles.

The sizes or envelopes for Split Expander (SE) and Full-Flow Staged Combustion (FFSC) engines, sized at 470k lbs. vacuum thrust and at the chamber pressures and nozzle area ratio's indicated, are shown schematically in Figure 2.3-8. These are shown in comparison with: (1) the SSME engine, and (2) STME/gas-generator engines sized at 580K lbs. thrust and at three different nozzle area ratio's as indicated.

	SSME (Ref)	Split Expander*	FFSC*
• Thrust (vac)	470K - 512K	470K	470K
• Chamber Pressure (psia)	3000	1200	3000
• Throttle Range	65% - 109%	75% - 100%	65% - 109%
• Mixture Ratio	6.0±	6.0	6.0
• Expansion Ratio	77.5	33.0	77.5
• Specific Impulse (vac)	453.0	436.3	453.0
• Engine Weight	6999#	4566#	5250#

Includes Installation Plumbing

* Note: Data From SRS Engine Cycle Analyses/Does Not Include Boost Pumps

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FIGURE 2.3-7 ENGINE CHARACTERISTICS DATA

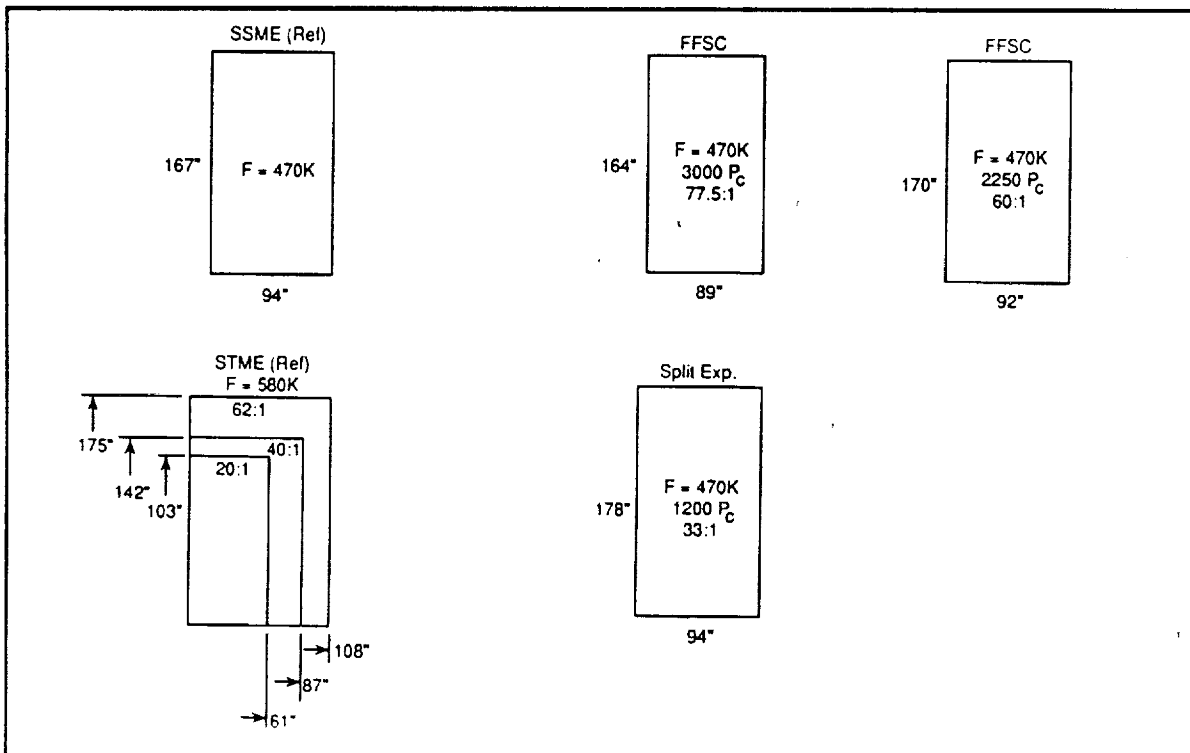

FIGURE 2.3-8 ENGINE ENVELOPE DATA

2.3.2.2 Engine Envelopes and AMLS Engine Installations

Base area requirements for STME/gas generator engines in AMLS booster and orbiter stage applications were examined in the "Propulsion Evolution Study" task, in comparison with propellant tank sizes estimated for those two stages. Sketches from that study task are shown in Figures 2.3-9 and 2.3-10. We can make similar analyses for SX and FFSC engines as was made for STME gas generator engine installations.

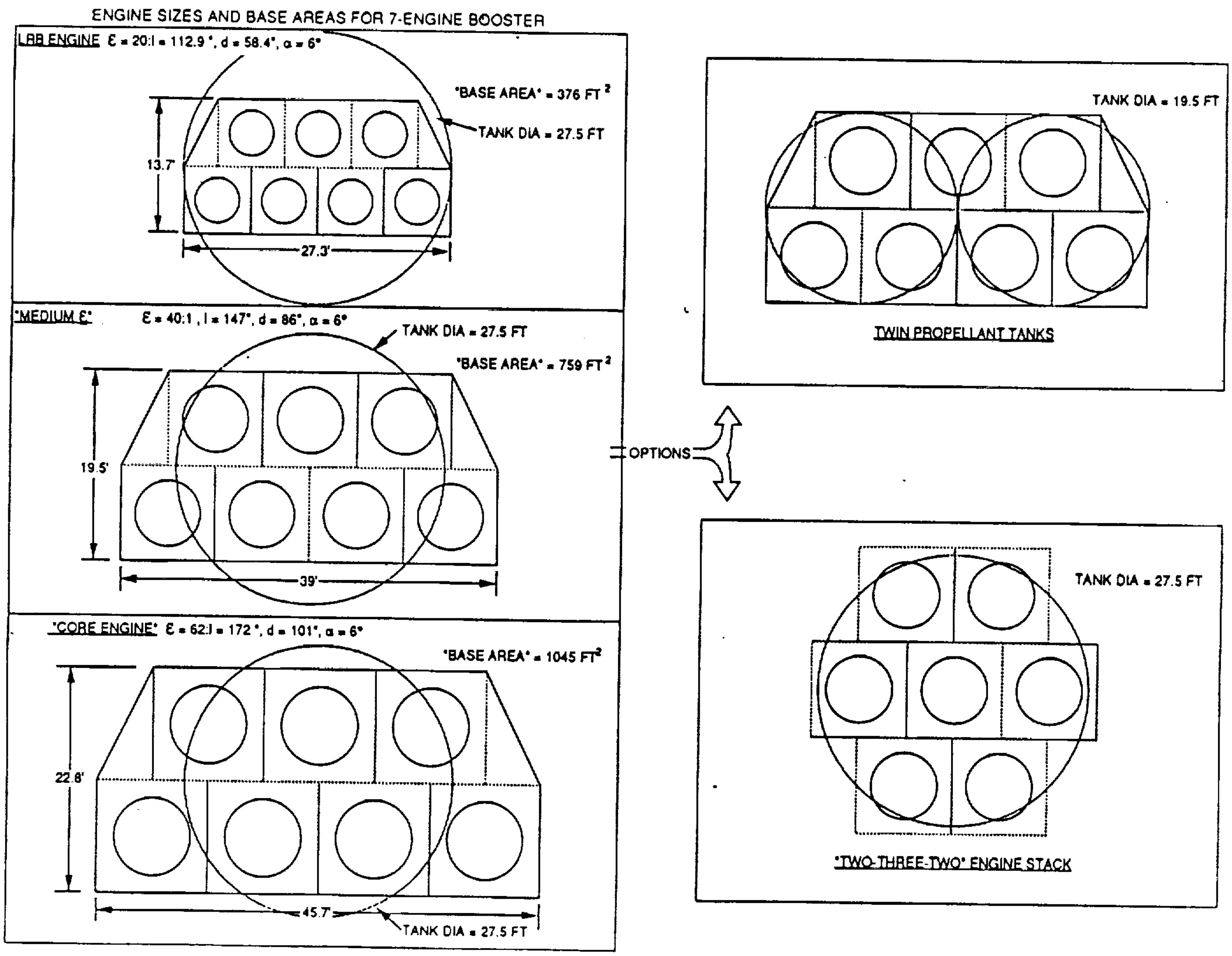

FIGURE 2.3-9 AMLS ENGINE SIZES AND BOOSTER BASE AREA

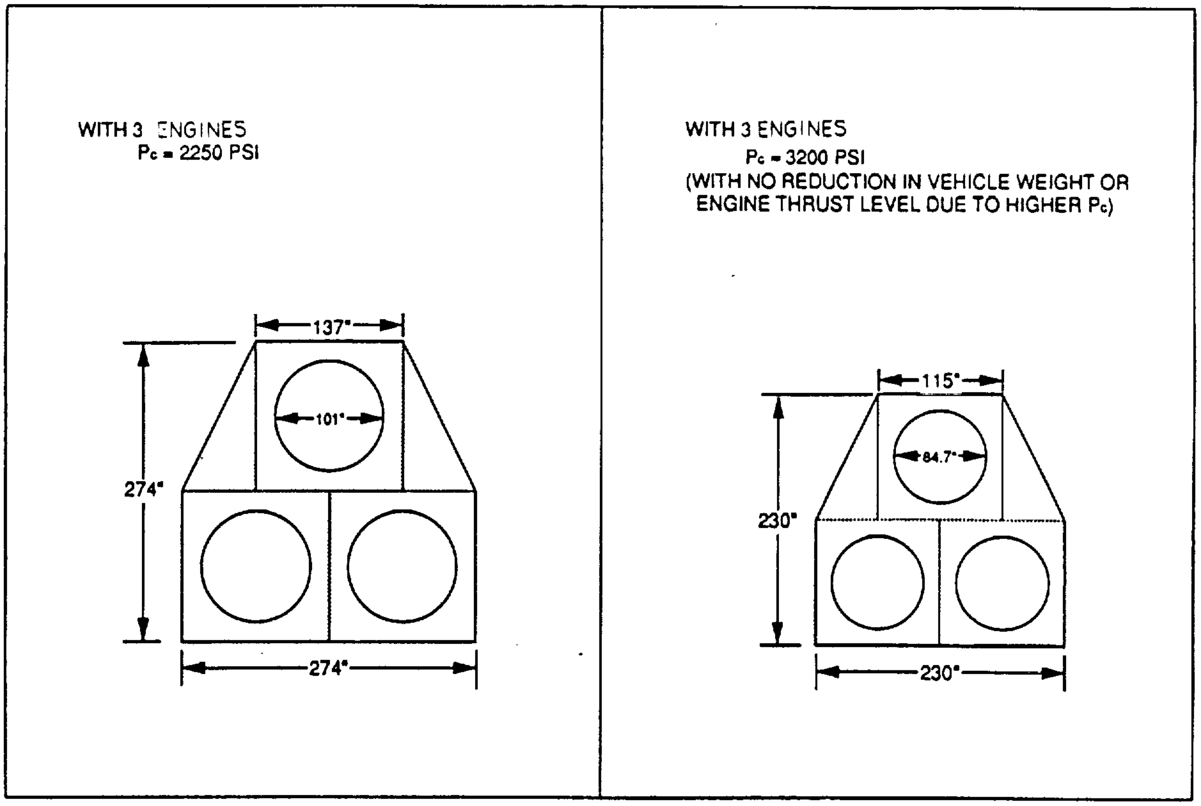

FIGURE 2.3-10 AMLS ORBITER ENGINE INSTALLATION

Analyses of STME/gas generator engines for the booster stages showed that engines with 20:1 area ratio nozzles would naturally be preferable from the standpoint of engine physical sizes; however, it appeared that engines with the ALS/STEP baseline 40:1 area ratio nozzles might be a workable fit. Three examples of possible engine and tank arrangements using 40:1 area ratio engines are shown in the figure. You will note in Figure 2.3-8 that the SE engine at 1200 Pc and 33:1 nozzle area ratio is slightly larger than the gas-generator (G-G) engine with 40:1 nozzle. An SE engine sized at the same thrust level as the G-G engine (580k lbs.) would be even larger, indicating that a nozzle area ratio lower than 33:1 would likely be needed for the

AMLS booster installation. The FFSC engine at a 3000 psi chamber pressure is quite similar in size to the SSME. It is very little larger than the 40:1 G-G engine in exit diameter, but is considerably longer due to the much higher nozzle expansion ratio (77:1). Although it might be possible to make the FFSC/77:1 engine size fit into an AMLS booster installation, there would be considerable motivation from these considerations to have a lower expansion ratio version for use in the booster stages. Earlier studies have shown a 35:1 nozzle area ratio to be favorable for SSME engines in booster applications; one would expect this to apply also for FFSC engines at similar chamber pressures.

Base area requirements for the orbiter stage with G-G engines at 2250 Pc and 62:1 area ratio are shown in Figure 2.3-10. The hypothetical example on the RH side of that chart shows the benefits in physical size that would accompany higher chamber pressures. The FFSC engine at 3000 Pc would provide a higher area ratio nozzle in a slightly smaller envelope, even if scaled up to the 580k lb. thrust level. The high chamber pressures and smaller physical sizes of the SSME and FFSC engines make them attractive candidates for orbiter stage installations. The Split Expander engine is on the other end of the chamber pressure scale (compared with SSME and FFSC engines), and would be larger than either the S-C or G-G versions at equivalent design values. If the SE example shown were scaled up to 580k thrust, the SE engine at 33:1 area ratio would be as large or larger than the 62:1 GG engine. This will obviously limit the use of this engine to area ratio's on the order of 33:1 (but not as low as current baseline values in ALS/STEP planning).

The engine spacing shown in Figures 2.3-9 and 2.3-10 include clearance for +/- 6 degree gimbal capability in both planes. With parallel mounted stages and large aerodynamic surfaces in vehicles such as AMLS, it is quite possible that more than six degrees of gimbal capability will be required (gimbal requirements for STS are +/- 8.5 degrees in yaw and +/ - 10.5 degrees in pitch). In that case, it would mean not only somewhat larger base area requirements, but addition of flexible sections of propellant feed ducting to allow this greater range of motion, with attendant increases in inert weight and installation space requirements.

2.3.2.3 Engine and Vehicle Performance

Basic engine data resulting from analyses of the Split Expander (SE) and Full-Flow Staged Combustion (FFSC) engines were shown earlier in Figure 2.3-7.

2.3.2.4 Split Expander Cycle Engines

A nominal data point resulting from SRS analyses of the SE engine cycle is shown in Figure 2.3-11, in comparison with SE data from other sources and in comparison with the baseline data for the gas-generator STME engine. The range of values from the STME studies (Aug 89 data) shows some spread in vacuum Isp, but a larger spread in estimates of engine inert weight. These are no doubt in part due to designing to different values for chamber pressure and nozzle expansion ratio. As noted earlier, two versions of the Split Expander engine are included in current "Space Transportation Engine Program (STEP)" definition studies and planning, in addition to the baseline gas-generator version. The first SE version is that indicated from ALS vehicle studies to be cost-optimum (1100 psia Pc), and the second represents a cost optimum from the engine viewpoint (900 psia Pc). Available data for these two points are also shown in Figure 2.3-11; corresponding data on engine weights are not yet available to us. It is assumed that that these STEP Phase B requirements data, for both SE and GG versions, are more nearly "spec values", and will generally run lower than "estimated values" from engine contractor and other analyses.

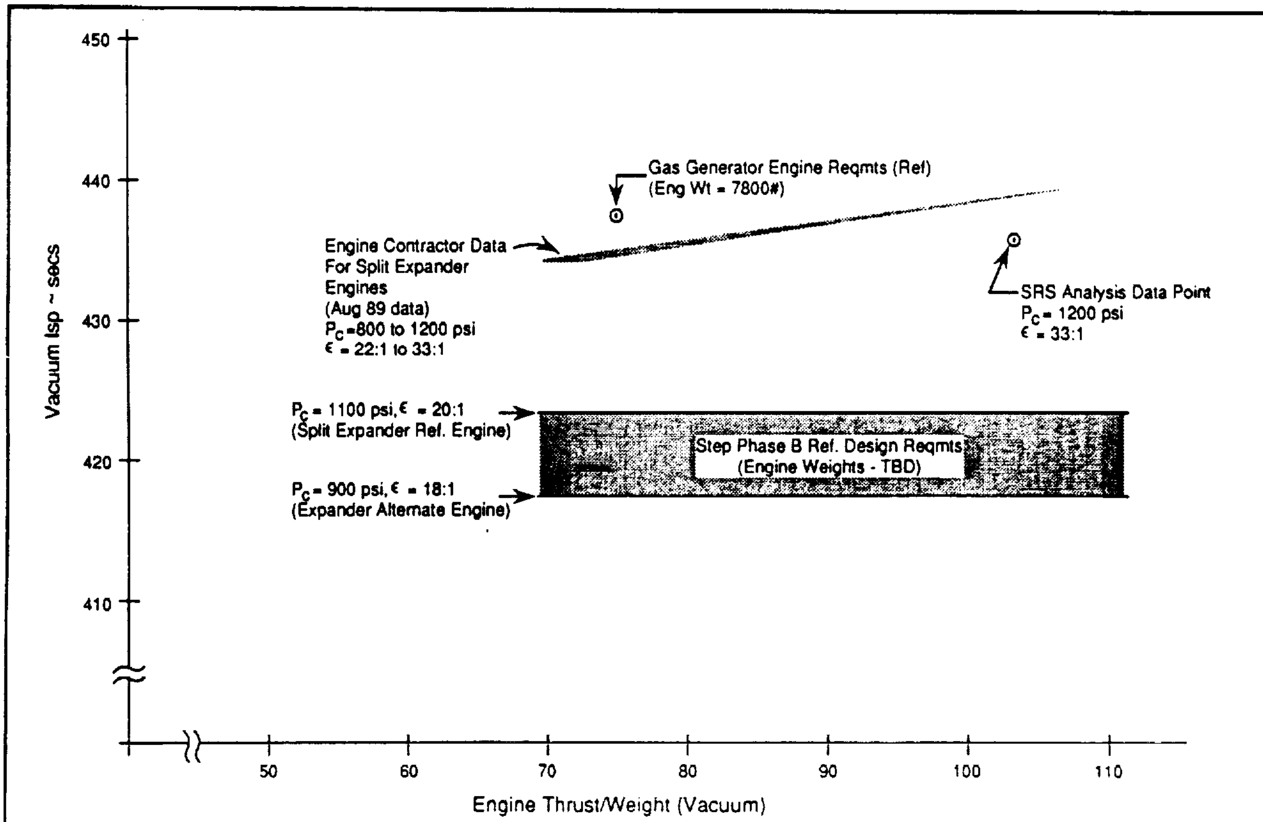

FIGURE 2.3-11 SPLIT EXPANDER CYCLE ENGINE COMPARISON WITH OTHER ENGINE DATA/ESTIMATES

Results from rough-order parametric sizing of AMLS type vehicles using SRS data/ characteristics for SE engines at 1200 Pc and 33:1 area ratio are shown in Figure 2.3.-12, in comparison with vehicles sized with SSME engines in both stages, and with vehicles sized with gas-generator versions of STME engines in both stages. This indicates the vehicle with SE engines to be approximately the same size/mass as the ones using gas generator STME engines. As a point of interest, a range of vehicle sizes resulting from use of SE engine data from the three STME engine contractor studies is also shown for comparison. Since vehicle gross weights for the cases shown in the figure are within a twenty percent range, no large discriminator is indicated. It is expected, however, that AMLS vehicles using engines designed to the current ALS/STEP baseline requirements, when these data become available, will show bigger differences in sizing for AMLS type vehicles.

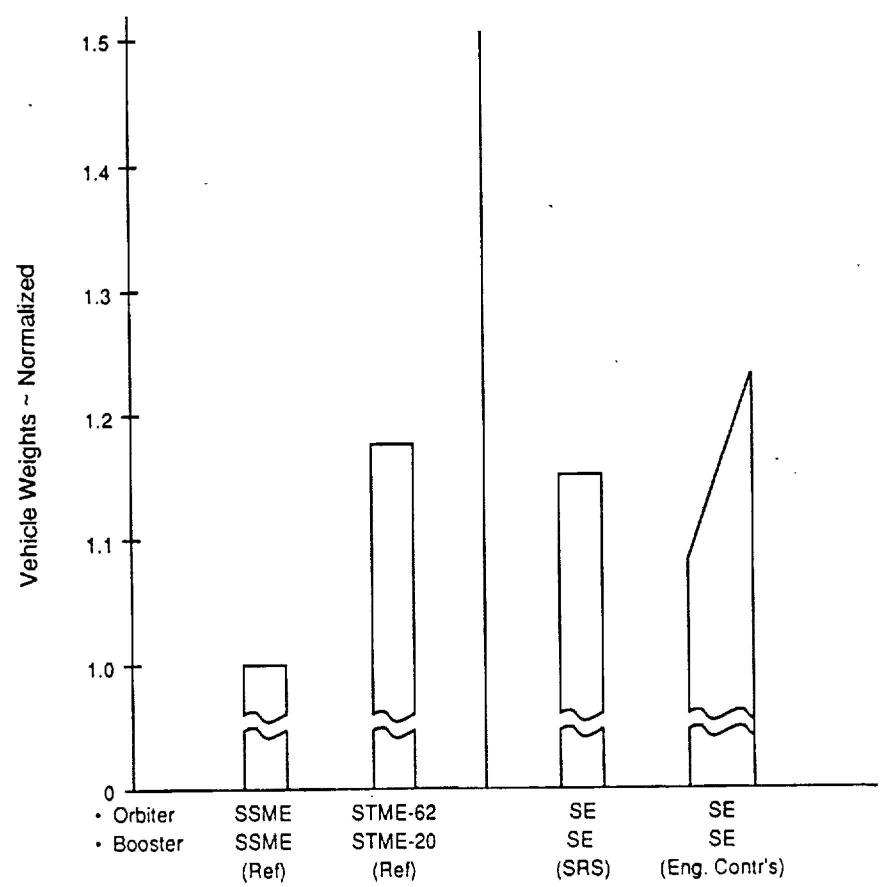

FIGURE 2.3-12 SPLIT EXPANDER CYCLE ENGINE AMLS (TSFR) VEHICLE CONCEPT WEIGHT COMPARISON

2.3.2.5 Full Flow Staged Combustion (FFSC) Engines

In this section, we will discuss briefly the sizing of AMLS type launch vehicles using FFSC engines; and, since one characteristic of FFSC engines is thought to be adaptability to operations over a range of mixture ratio's, we will examine some rough-order mixture ratio trades in two-stage fully reusable AMLS vehicle concepts.

Basic data for FFSC engines designed to operate at mixture ratio's of 6:1 to 9:1 are shown in Figure 2.3-13, as an expansion of the basic data provided for the 6:1 ratio version shown earlier in Figure 2.3-7.

AMLS vehicles sized with FFSC engines are shown in Figure 2.3-14, showing the gross weights of these vehicles in comparison with reference vehicles sized with SSME engines in both stages, and with vehicles using gas-generator STME engines in both stages. Not surprisingly, with the similarity between FFSC engine data and SSME engine data, the vehicle sizes are not far different, with both being lower in weight than vehicles sized with either gas-generator/STME engines or Split Expander engines.

		Engine Mixture Ratio			
		6:1	7:1	8:1	9:1
• Vacuum Thrust	(lbs)	470K			
• S/L Thrust	(lbs)	378.5K	380.5K	380.9K	381.3K
• Chamber Pressure	(psia)	3000			
• Expansion Ratio		77.5:1			
• Specific Impulse - Vac	(secs)	453.9	449.4	435.6	421.8
• Specific Impulse - (S/L)	(secs)	365.5	363.8	353	342.2
• Engine Weight	(lbs)	5899	5695	5654	5645

FIGURE 2.3-13 FFSC ENGINE DATA AT SEVERAL MIXTURE RATIOS

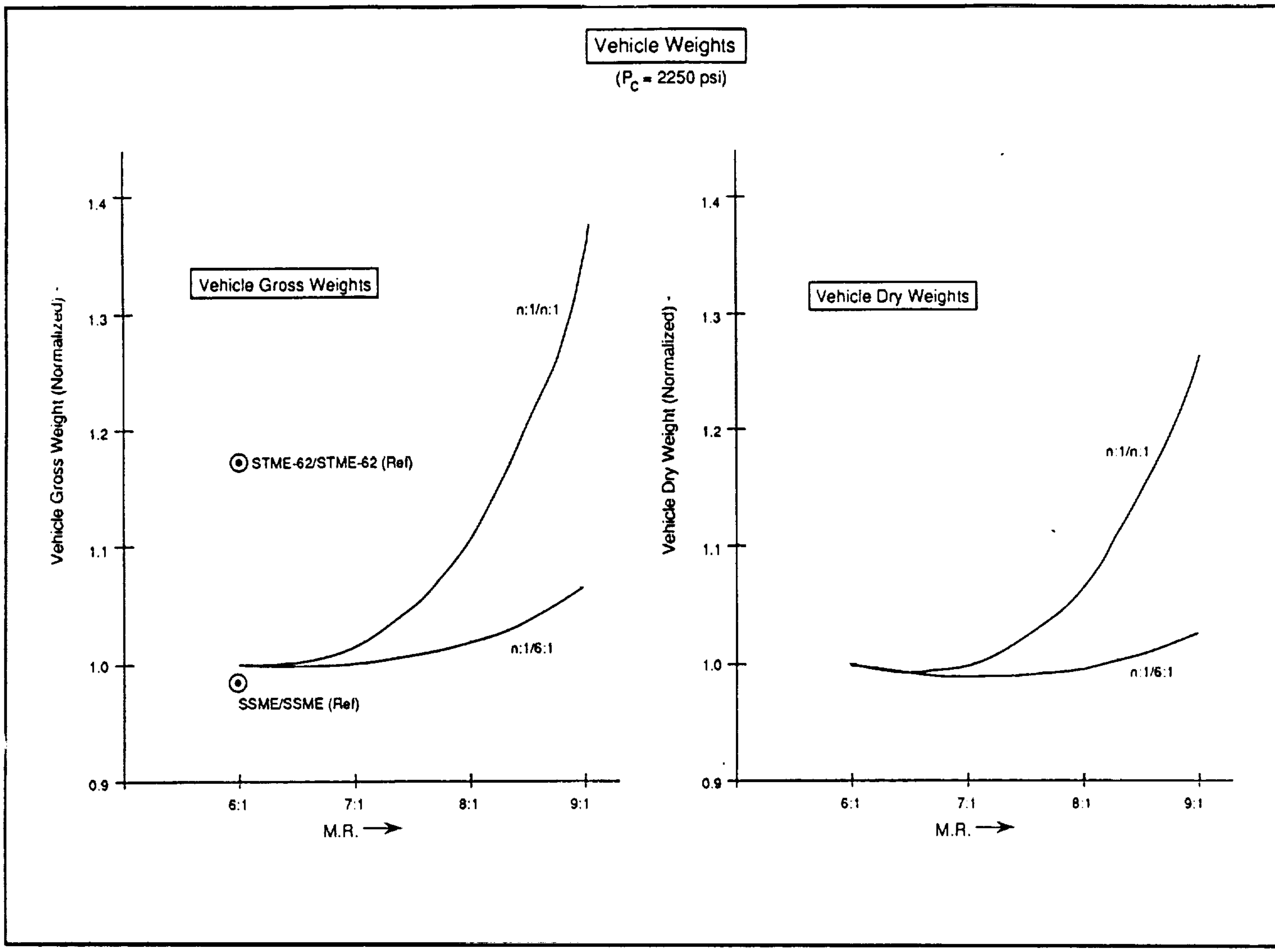

FIGURE 2.3-14 FFSC ENGINE MIXTURE RATIO TRADES FOR AMLS VEHICLES - WEIGHTS (PC = 2250 PSIA)

Results from engine mixture ratio parametric trade studies based on FFSC engines at 2250 psi chamber pressure are also shown in Figure 2.3-14, in one case varying mixture ratio for the booster stage while holding the orbiter mixture ratio constant at a value of 6:1 ("n:1/6:1"), and in the other case varying mixture ratio at the same value in both stages ("n:1/n:1"). As in several other similar trade studies, this analysis indicates a fairly flat curve for variations in booster mixture ratio ("n:1/6:1"), with possibly a slight "bucket" in vehicle dry weights in the 7:1 to 8:1 range. However, a distinction that fine would be beyond the accuracy level of these analyses. Varying mixture ratio's in both stages simultaneously ("n:1/n:1") indicates an increasing disadvantage in weights with higher mixture ratio's.

As a point of interest, variations in propellant tank volumes are shown in Figure 2.3-15, corresponding to the vehicle gross weight and dry weight variations shown in Figure 2.3-14. The LH side of the figure shows a significantly lower booster propellant tank volume at a mixture ratio of 7:1, compared to the baseline value of 6:1. It might be expected that there would be a larger corresponding difference in vehicle hardware/dry weights than indicated in the preceding figure. Hopefully, further analyses of these trades can pursue whether this might be the case.

Mixture ratio trades were also run using data for FFSC engines designed at 3000 psi chamber pressures, to see if this might show a more pronounced trend. This turned out not to be the case. As shown in Figure 2.3-16, the gross weight and dry weight curves for the "n:1/6:1" case are equally as flat as in data using FFSC engines at 2250 psi chamber pressures. As might be expected, these data show lower vehicle weights, resulting from improved engine performance at the higher chamber pressure. As one further "sensitivity study", this trade was repeated with significantly higher booster stage delta-v's (1.5 to 2 times the nominal value), to see if influences of mixture ratio variations might be more pronounced. The results, however, showed no major differences. We have not yet in this study task examined vehicle applications with engine mixture ratio's varied during flight. We anticipate that this trade will be included as a part of follow-on studies of undeveloped engine cycles.

2.3.2.6 Summary/Observations

At the level of "granularity" of these vehicle applications analyses, we will see primarily the effects of engine characteristics on vehicle performance and sizing, and the considerations of engine physical sizes in relation to vehicle installations. Due to their design for return/reuse and airplane-like flight and landing characteristics, the AMLS class of vehicles will be generally more sensitive than other vehicle classes to engine performance characteristics and to the physical size of the engines.

Engine performance characteristics and physical sizes would naturally be favorable for SSME and the Full-Flow Staged Combustion (FFSC) variant of the same basic engine cycle. The SSME or FFSC engines with nozzle area ratio's close to that of the current SSME would be very favorable for AMLS orbiter installations. The same basic engines with reduced area ratio nozzles (on the order of 35:1) would adapt better to the AMLS booster installation.

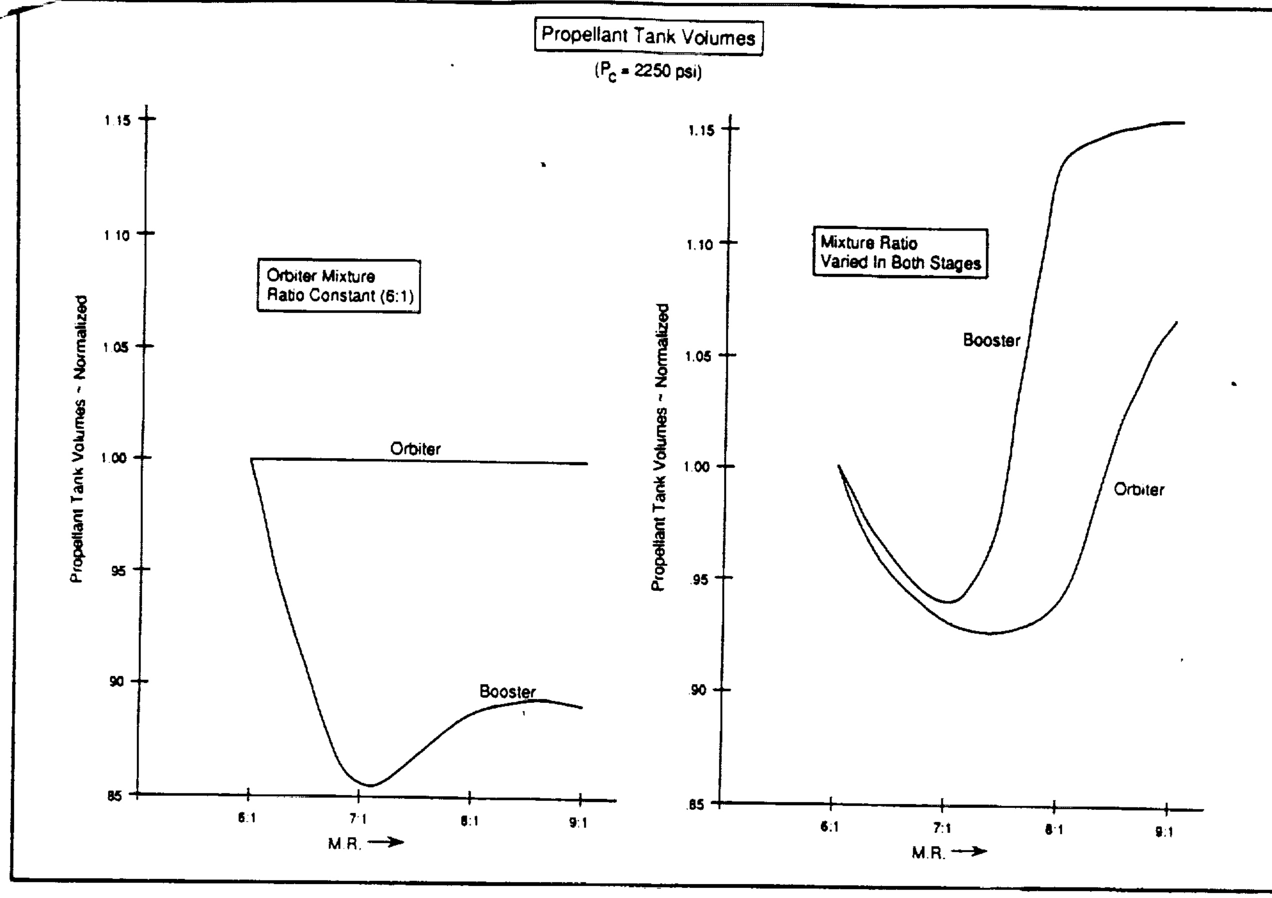

FIGURE 2.3-15 FFSC ENGINE MIXTURE RATIO TRADES FOR AMLS VEHICLES - TANK VOLUMES (PC = 2250 PSIA)

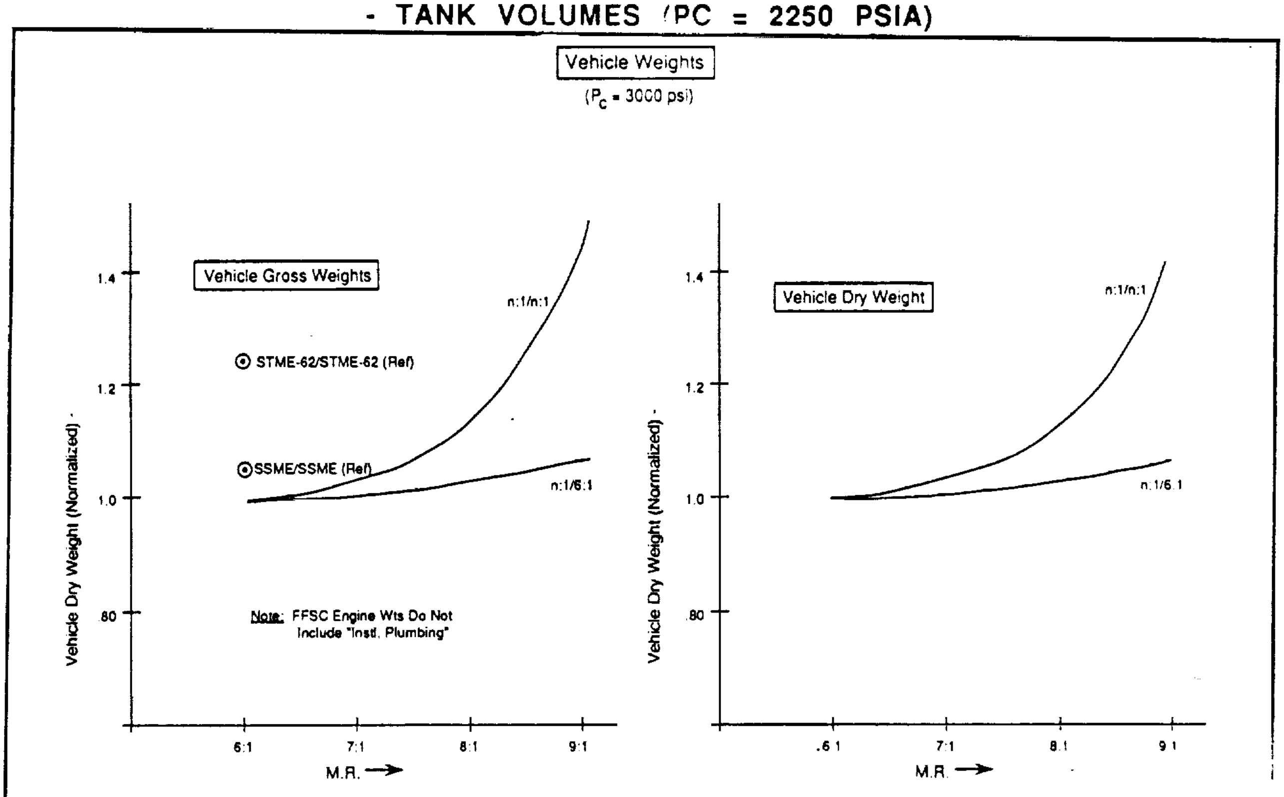

FIGURE 2.3-16 FFSC ENGINE MIXTURE RATIO TRADES FOR AMLS VEHICLES - WEIGHTS (PC = 3000 PSIA)

At the design parameters used in this study for the Split Expander (SE) cycle engine, vehicle performance is indicated to be competitive with vehicles equipped with gas generator versions of STME engines (Pc=1200 psi and 33:1 area ratio). However, the larger characteristic physical size of these engines would likely require use of a lower area ratio in the booster stage, and would preclude going to higher expansion ratio's for the orbiter application. When data become available on engines designed to the current STEP Phase B requirements, it is expected that these data will show reduced performance in AMLS vehicles and significant increases in vehicle sizes.

Trade studies comparing operation of FFSC engines over a range of mixture ratio's did not indicate any strong advantage for mixture ratio's different from the current baseline value of 6:1. Some indications of an inert weight minimum at mixture ratio's of 7:1 or 8:1 should receive further study. With the higher inert weight characteristics for AMLS type vehicles, one would expect them to be more sensitive to propellant tank volume requirements than ALS or PLS type vehicle concepts.

Considerations other than engine/vehicle performance, physical sizes and engine installation considerations must obviously be included considering the pro's and con's of these candidate engine cycles, and several of these factors have been examined in the previous part of this subtask. Reductions in severity of operating environments for some of its major components, or striving for simpler engine designs and higher inherent engine reliabilities, would obviously be of direct benefit in manned vehicle applications, if realized. Emphasis on reduced unit costs for engines would benefit any launch vehicle application, but would be of lesser motivation for highly reusable launch vehicles such as AMLS, than in cargo launch vehicles where engines are expended or reused a few times.

2.4 Technology Development Requirements (Subtask 4)

2.4.1 Split Expander Cycle

Issues related to split expander cycle technology development were discussed in Section 2.3.1.1. A forthcoming contracted effort, the Split Expander Cycle Demonstration Program" (MSFC), will demonstrate the feasibility of a 600 Klb split expander cycle engine using a Thrust Chamber Assembly designed especially to demonstrate the cycle. The RFP states that the program will concentrate on major technological issues and will be conducted in two phases: 1) a preliminary design phase, and 2) a detailed design, fabrication, and test phase.

Another contracted effort (at LeRC) to design and build a demonstration model of the split expander engine with thrust levels up to 50 Klb is also expected. The current status of the program indicates that it has not kicked-off yet, but has been announced and is being negotiated. Technology transfer from this program will likely be instrumental in the organization and planning of a full scale 500K split expander program. The MSFC split expander demo engine program will not be of flight quality, but will provide many "lessons learned".

The RL10 (expander cycle) has a proven restart capability with no boost pumps. Technology must be further developed to support any potential restart problems encountered due to the complexities of having a larger engine (thrust, weight, and geometry), and difference between the split expander and pure expander cycles (e.g., additional valves and control systems). However, there should be no boost pump technology development required, since there will probably be no boost pumps.

There will be some manufacturing processes that will need to be developed for some engine peculiar parts. However, the relatively low operating pressure and temperature significantly simplifies the complexity and makes allowance for large margins easier. The integrated flexible feed system (wrap around ducts) is one such engine/vehicle peculiar part which may need to be developed depending upon vehicle applications. Cast nozzle and turbopump housings are other examples.

Safety and reliability issues should pose no major technology development. Again, the low chamber pressure and temperature inherently makes this engine relatively safe. The low chamber pressure also makes possible the use of non-exotic materials which are cheaper and easier to use.

2.4.2 Full Flow Staged Combustion Cycle

The full flow cycle offers certain advantages over both gas generation and dual fuel-rich preburner staged-combustion cycles. When assessing the performance issues relative to the FFSC cycle there are several key advantages. Because the FFSC is a topping cycle, it can attain the maximum I_{SP} for a given set of design conditions (P_C, MR, ε, etc.) as compared to other cycles. The gas-gas main injection feature of the FFSC has the potential for high C^* and low L^*, stable combustion, and a wide throttle range (due to constant volume injection) as compared to other cycles.

In the manufacturing area, there are also several key advantages. Because turbopump efficiency is not critical, due to large turbine flow rates, large tolerances and simple but rugged blade designs are permitted. The low LOX turbine inlet temperatures as a result of the high LOX turbine flow rate of the full flow concept, also allows for wider materials choices and less critical quality assurance standards (i.e., larger materials flaws could be permitted). The FFSC cycle is also ameamable to axisymmetric turbopump housings.

Operationally, the full flow concept has several key advantages. Because the fuel and oxidizer turbopumps both run rich on the pumped propellant (i.e., fuel side LH_2-rich and oxidizer side LOX-rich) the system better lends itself to throttling and high/variable mixture ratio operations due mainly to the avoidance of proportional parasitic losses associated with running gas generators and preburners LH_2-rich (or both LOX-rich). Given the FFSC capability of high/variable mixture ratio engine operation, a common engine could be developed for both booster (high mixture ratio for high ρ I_{SP} performance and core/orbiter, lower mixture ratio for high I_{SP} performance) applications. In addition, given the LOX-rich turbine gasses for powering the LOX pump, the criticality of drive gas-pumped propellant separation on the LOX pump side, as compared to the LH_2-rich turbine drive gas on the SSME LOX pump, is greatly reduced and in fact inert gas purges on the seals separating the LOX turbine and pump will probably not required. Gas-gas main chamber injection may also eliminate pogo suppression devices because of the inherent compressible nature of gas-gas injection and the damping of low frequency pressure fluctuations that might propagate back through the liquid in the propellant feed system. The compressible nature of gas-gas injection also makes for a softer start.

The reliability and safety areas for the FFSC cycle also exhibit definite advantages also. Bearing lifetime, as current concern in both the current SSME LOX pump and probably in the SSME alternate turbopump designs being configured for a LH_2-rich turbine drive, would benefit from the lower bearing DN, loading, and cooling requirements associated with the full flow concept which requires substantially lower turbine inlet temperatures. These lower temperatures, stresses, thermal gradients/transients could result in the utilization of lower cost materials and/or higher thermo/structural margins. A potential for a fewer number of components and auxiliaries also exists including a single stage LOX impeller and turbine, simple shaft seals (because the criticality of LH_2-rich drive gasses and LOX mixing in the LOX turbopumps is eliminated). This also could lead to integral turbopump assemblies (TPA's), and preburners and valves which might also improve reliability. Neither the criticality-1 LOX heat exchanger as configured for the current SSME nor the new external heat exchanger concepts would not be needed for autogenous LOX tank pressurization, since these gasses would be produced by the LOX turbopump preburner. The softer start characteristics of gas-gas main injection would also allow verification of engine operation during throttle-up and a safe functional cut-off potential during start. The larger LOX turbopump operating clearances possible because turbopump efficiency is not as critical for the FFSC cycle would also improve reliability. The smaller quantity of propellant in the lines and manifolds since most high pressure propellants are in the gaseous phase, would also tend to increase safety. The elimination of several criticaility-1 failure modes (LOX pump bearings, heat exchanger, etc.) would also result in safer engine operations. The flexible nature of the FFSC cycle (such as high/variable mixture ratio operations) is further enhanced by its independent TPA operations allowing for ease of TPA clustering and permitting a redundant TPA philosophy to be explored to increase reliability as opposed to a redundant engines philosophy for "engine out" to increase reliability.

Many of the features/characteristics of the FFSC cycle could lead to significant cost savings. The lower temperature, lower stress environments in for example the LOX turbopump and main chamber injector, would permit a wide selection base of materials with good physical properties, resulting in low materials costs. In addition, given these lower temperature/stress environments, more simplistic designs and resultant reductions in critical quality assurance requirements, lower cost forgings/castings could be used. The reduction in auxiliary components (such as LOX Heat Exchangers) could also result in lower cost engine configurations as compared to conventional staged-combustion cycles (SSME) and gas generator cycles (STME).

In order to develop a full-flow staged-combustion cycle engine, several technology development programs must be initiated and successfully completed to mature the subsystem/component technology sufficiently for full-scale development. Technology developments are described in the following subsections.

2.4.2.1 Oxidizer-Rich Preburners

The LOX turbopump in the full flow cycle is driven by an oxidizer-rich preburner. Running at a mixture ratio of approximately 180, this preburner provides gaseous oxygen to drive the turbine. As illustrated in Figure 2.4-1, a typical preburner may be annular in order to integrate effectively into the LOX turbopump. As described in Section 2.3.1.2, the preburner is to be designed with a localized combustion zone, submerged in a diluent flow to protect the chamber walls and injector face. Promoted mixing below the combustion zone will be used to produce a uniform mixed mean temperature of gas entering the turbine.

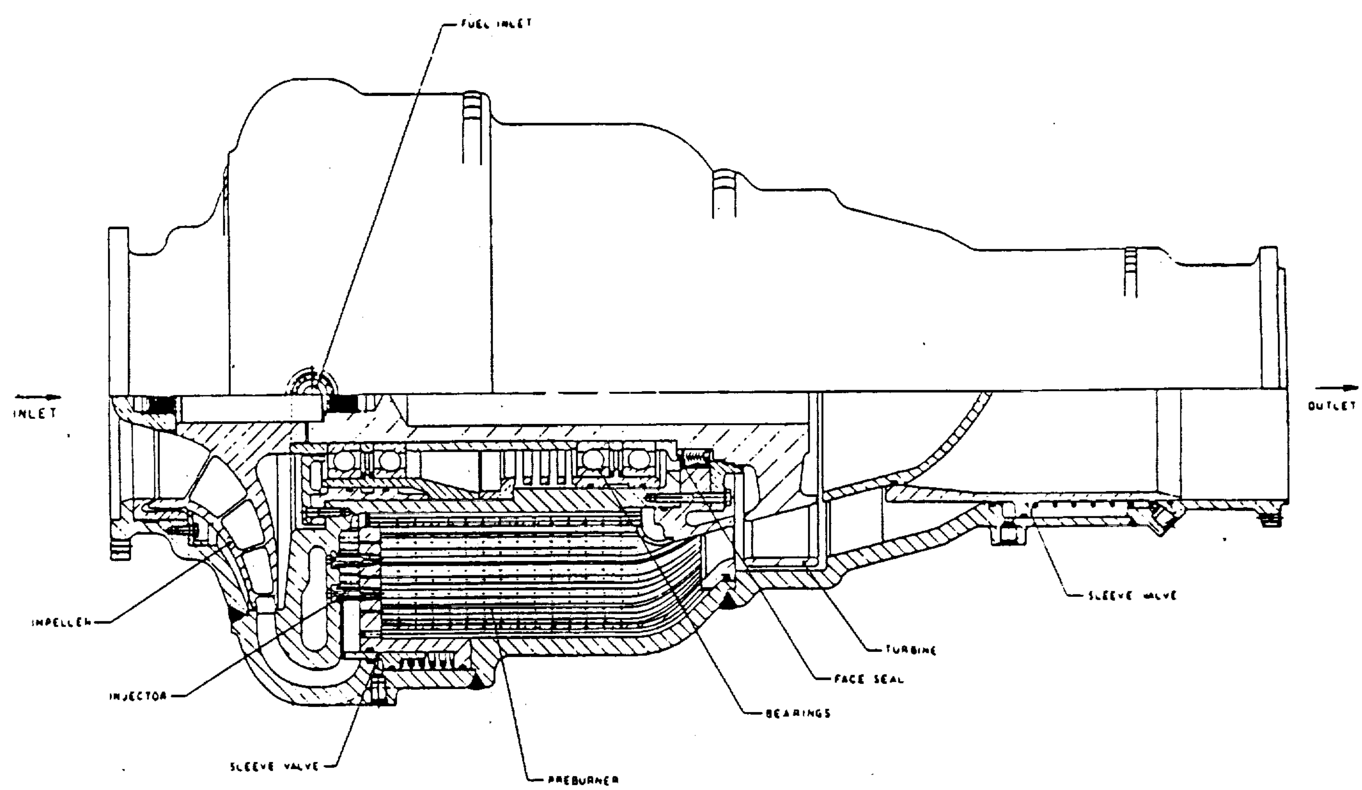

FIGURE 2.4-1 FFSC LOX TURBOPUMP

Current state-of-the-art for high mixture ratio oxygen/hydrogen preburners is illustrated in Figure 2.4-2. This workhorse LOX/LH_2 combustor was designed and run by MSFC at mixture ratios from 20 to 150 at 1000 psig chamber pressure. The chamber was copper with a water-cooled throat. The injector was a coaxial configuration in a rectangular pattern. Test operations over the entire mixture ratio range were successfully conducted by NASA MSFC in 1965.

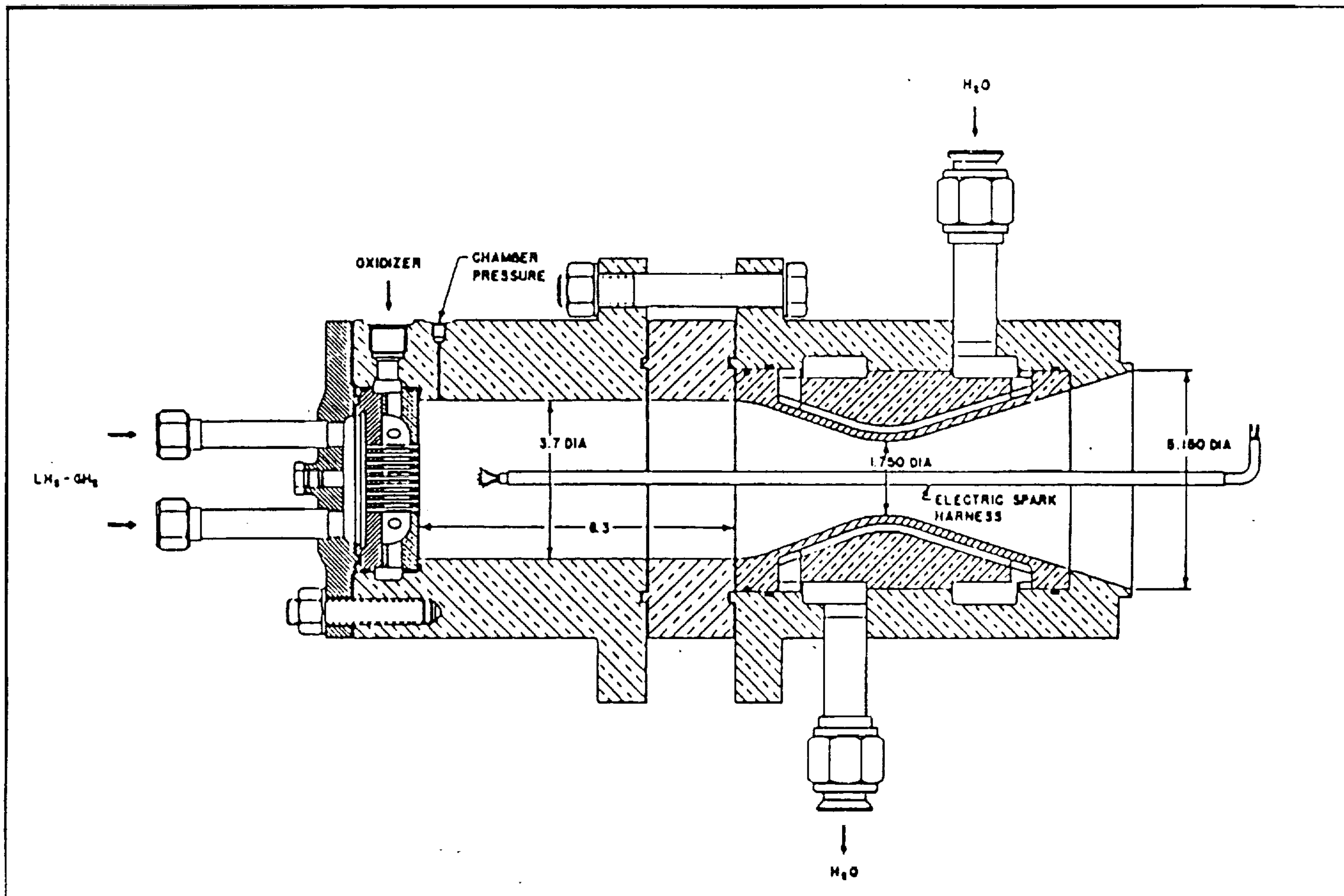

FIGURE 2.4-2 HIGH MIXTURE RATIO PREBURNER

2.4.2.2 Full-Wall Main Propellant Injection

In anticipation of the rapid complete combustion possible with gas-gas injection, it is contemplated that L* of the main chamber could be reduced to a low value (on the order of 20-40). The small chamber surface area for such a volume would permit propellant injection over virtually the full chamber wall down to the region of convergence near the throat. The advantage of this concept is that the chamber is essentially gas transpiration cooled by the main propellant injection and the chamber and injector become one and the same part so the expense of two separate components is avoided. Figure 2.4-3 illustrates this concept and also reveals that a relatively simple, low cost design may be practical.

Current state-of-the-art for this concept is that CFD programs for internal thermal and ballistic analyses of combustion chambers and rocket nozzles are available to analyze and predict the combustion, thermal and structural characteristics of such a chamber. Suitable chamber materials are available, and the chamber can be manufactured by conventional processes.

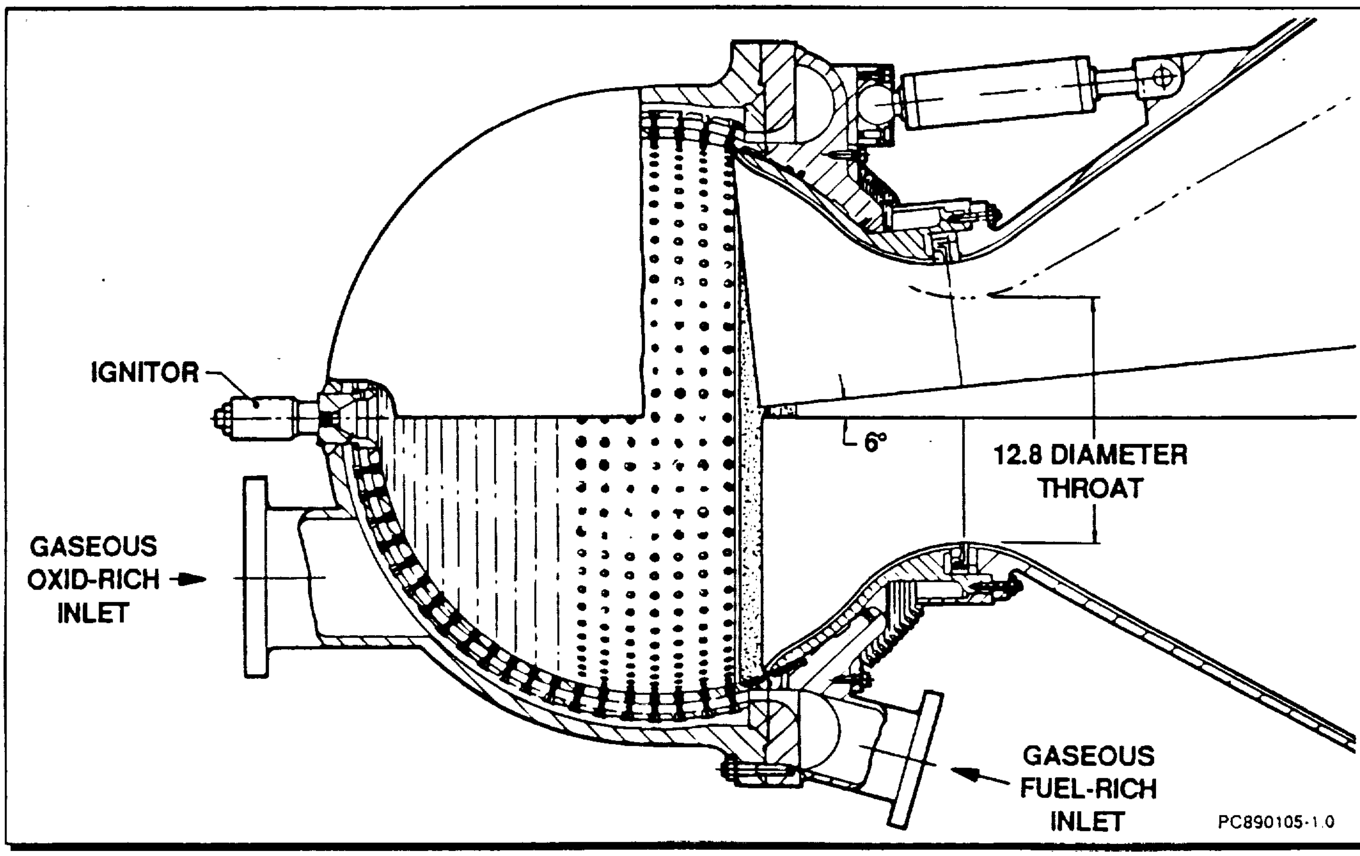

FIGURE 2.4-3 FULL WALL INJECTION CHAMBER

2.4.2.3 Pre-Mix Main Injector

The injector for the full flow cycle engine is perceived as having discreet oxidizer elements injecting GOX into a "cloud" of hydrogen-rich gas to burn. Mixing would occur concurrently with combustion near each local injection site. This may be entirely satisfactory.

However, since the combustion products in the shear layer between the two propellants may tend to expand and isolate the unburned propellants from further mixing/combustion, an alternative is considered.

The gas-gas injector provides the opportunity to pre-mix the gases prior to injection into the chamber in a manner analogous to a gas welding torch. Figure 2.4-4 illustrates the pre mix concept. The advantage of this innovation is that complete mixing of the gases is achieved prior to injection so that rapid, complete combustion at a precise mixture ratio can be achieved.

The gases mix in a porous metal fiber mesh which ensures mixing via tortuous common flow paths. Also, the presence of the metal fiber mesh acts as an ignition preventative in the same manner that meal screens in gasoline tanks effectively prevent ignition of the mixture of the hydrocarbon vapors and air.

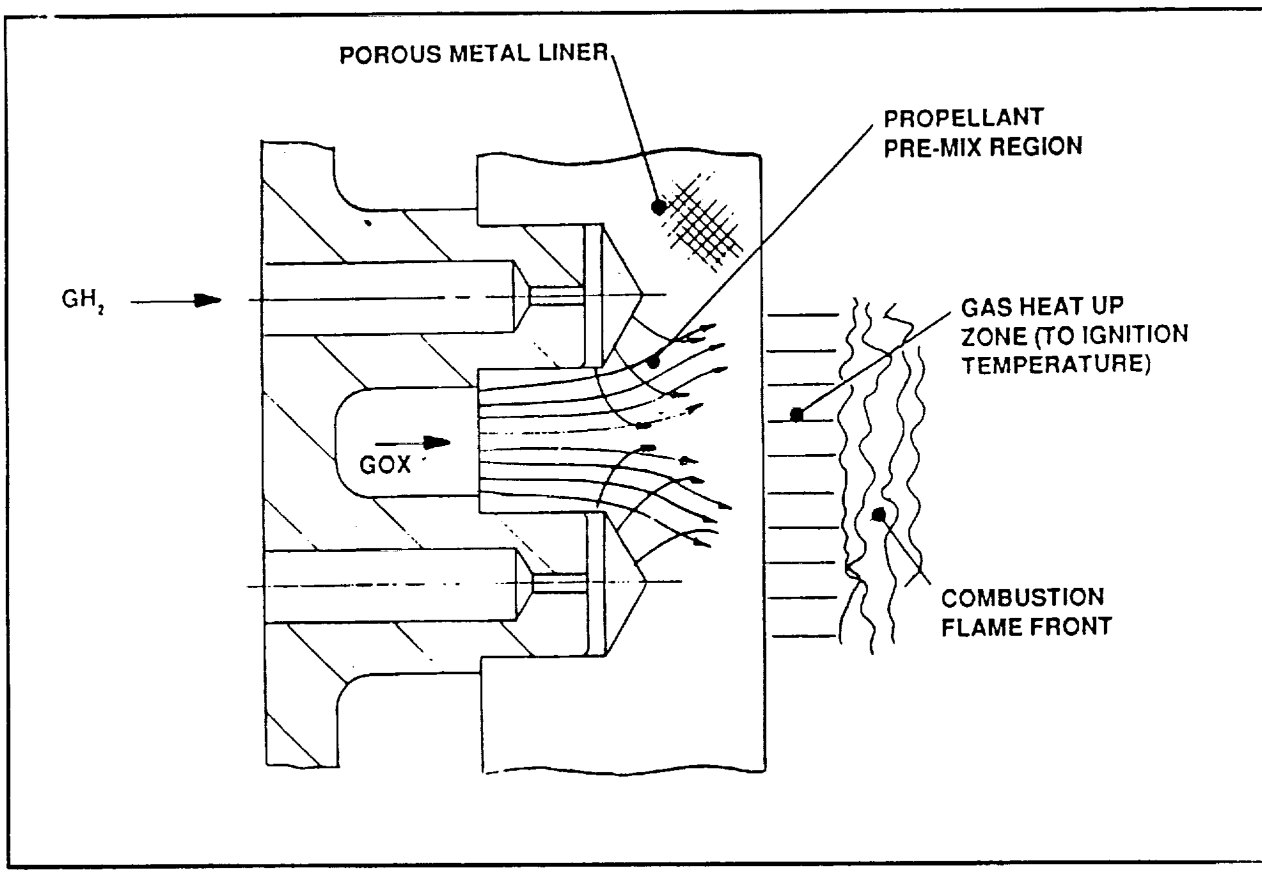

FIGURE 2.4-4 PRE-MIX INJECTOR CONCEPT

The through-flow of cool expanding gases acts to keep the porous metal cool, removing the heat transferred from the nearby flame front. Injection velocity profile is chosen so that the flame front stands off the surface even at the deepest throttle setting.

2.5 Technology Development Program Plan (Subtask 5)

The current split expander cycle demonstration program is addressing technology development requirements for split expander cycle engines for booster propulsion, and we will not assess, evaluate, or duplicate their program planning. However, we will define a Technology Development Program Plan for the FFSC engine.

A technology development program is outlined to investigate, validate and demonstrate the technologies discussed in Section 2.4.2. The program suggested spans a 6-year period, and contemplates total funding of 4.442M ranging from slightly over .5M in some years to just under 1.0M in other years. Each of the early years is arranged to validate one technology, which could be than pursued as a separately funded project. If all the technologies were to be fully pursued, the overall program could produce a new engine in six to eight years, depending on the thrust level chosen. This engine would not require technology breakthroughs, and could be readily developed to early maturity with very low development risk. The resulting engine has high potential for meeting the cost, safety and reliability goals needed for routine access to, and safe operation in space.

2.5.1 FFSC Technology Development Tasks

2.5.1.1 Engine System Definition

The purpose of this task would be to conduct engine system engineering efforts. This work will provide upper level specifications, design and operational requirements, sizing and interface guidance, and; in short, provide the general engine system definition framework to guide and coordinate other tasks of the program. The benefit of this effort is that the other tasks will have a common baseline engine to use as a reference. This will result in task end items which will be compatible for integration as they are demonstrated.

This effort also incudes the detailed analyses and reporting of test results from other tasks. Design documentation and configuration management will also be provided under the engine system definition function.

2.5.1.2 Oxidizer-Rich Preburner

The objective of this task would be to demonstrate the technology of oxidizer-rich combustion. Conceptual design bases have already been prepared for several combustors, ranging from small units for generating ullage pressurant gas, to special combustors for use as materials testers, and to configurations specifically designed for use as turbopump drive gas producers.

In this task, a concept will be chosen for development and demonstration testing. The basis for the choice will be that the unit has end-item utility, can be testing in existing facilities, and provides a significant data base in oxidizer-rich combustion technology verification.

Testing will commence in the fifth quarter of the program, and envisions the use of NASA facilities at MSFC. The testing will demonstrate operation over a range of mixture ratios from 20 to approximately 200, using LOX/LH_2.

2.5.1.3 Pre-Mix Gas-Gas Injector

The objective of this task is to demonstrate the pre-mix gas-gas injection concept. The work will include laboratory scale bench tests of the concept and extend to short duration firings of small heat sink chambers at up to 2000 psia chamber pressure.

It is envisioned that this effort will be conducted at NASA facilities for both the laboratory scale tests and the heat sink chamber firings. Acurex proposes to provide the engineering effort, hardware, and test support, with NASA providing facilities and facility operations.

2.5.1.4 Full Wall Injection Chamber

The objective of this task is to demonstrate the concept. There are two key considerations. One is injector design for assurance of cooling, the other is chamber geometry that will hold a stable combustion zone flow distribution. CFD modeling will be used to define the design bases for both of these aspects. Testing will include initial cold flow tests for flow visualization and verification of CFD modeling. Testing will proceed to ignition tests and short firings at low chamber pressure and cool mixture ratios for further verification of modeling and facility operations. Testing will progressively move to firings of up to fifteen seconds and chamber pressure of 2000 psia in a 5K-10K lb thrust chamber size.

2.5.1.5 Integrated Engine Testing

The overall technology program could result in a set of matched test hardware, which could be assembled and tested, (pressure-fed) with other necessary subsystems (i.e., fuel-rich preburner, nozzle, etc.) to demonstrated all the major combustion devices of the engine.

This task has the objective of integrating the hardware from the prior tasks, adding the fuel-side propellant supply and nozzle, and firing the overall assembly. The preprototype engine would provide proof of principal for a family of very low cost propulsion systems.

2.5.2 Program Schedule

The program schedule is shown in Figure 2.5-1. It summarizes the elements and time-phasing of the respective tasks. The program has been arranged to permit the hardware produced in one task to be used in subsequent tasks to maximize benefits that can be achieved. However, it is not a requirement that the tasks be so integrated, and tasks can be scheduled in other sequences if desired.

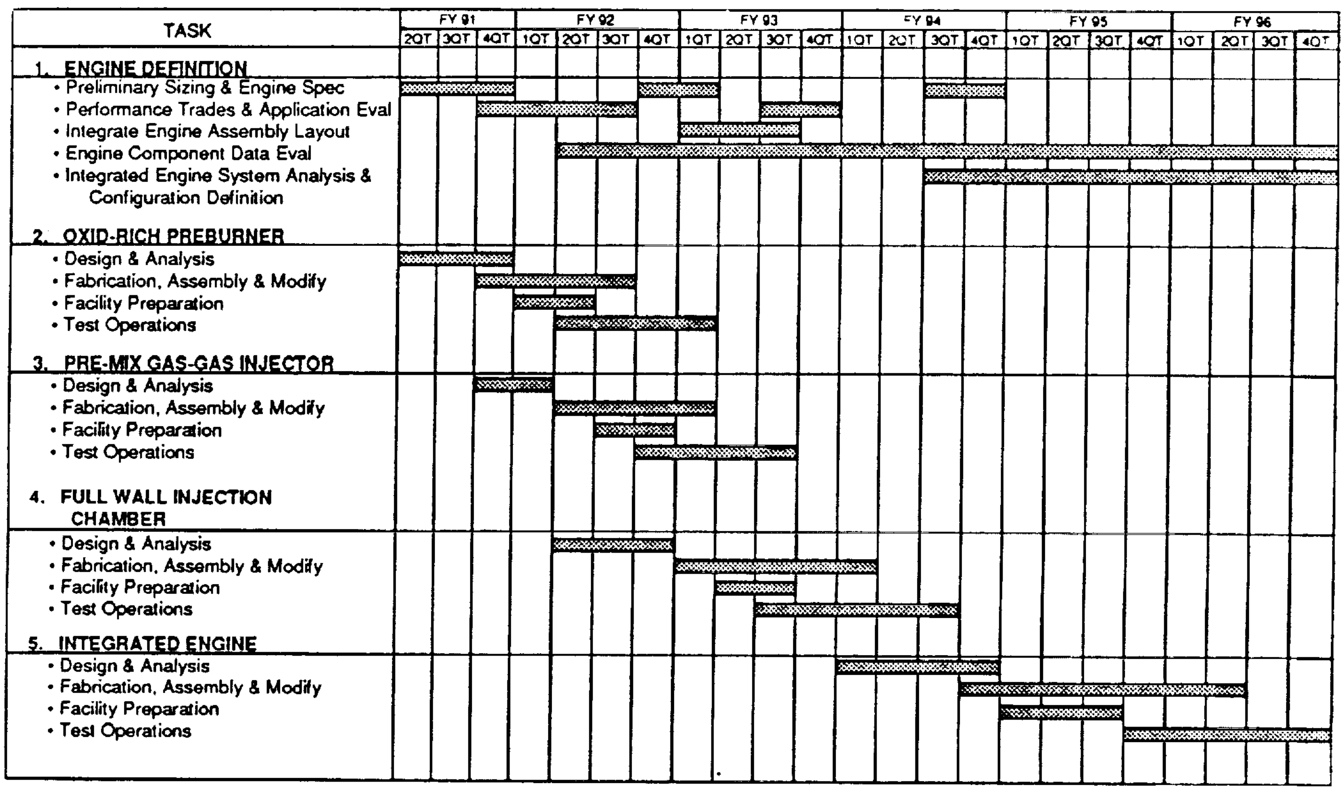

FIGURE 2.5-1 FFSC TECHNOLOGY DEVELOPMENT PROGRAM SCHEDULE

2.5.3 Program Costs

Table 2.5-1 shows a budgetary cost spread for the overall program. The costs shown include engineering labor and test hardware. No facilities or test operations costs are included under the assumption of GFE facilities and test consumables. The costs shown are for a task sequence which tends to minimize cost for the overall program, since many of the tasks are mutually supportive.

TABLE 2.5-1 FFSC TECHNOLOGY DEVELOPMENT PROGRAM COSTS

LABOR	FY 1991	FY 1992	FY 1993	FY 1994	FY 1995	FY 1996	TOTALS
COSTS	$465,000	$674,000	$594,000	$518,000	$671,000	$530,000	$3,452,000
FABRICATION COSTS							
PREBURNER	$60,000	$60,000					$120,000
INJECTOR		$180,000					$180,000
CHAMBER			$200,000	$50,000			$250,000
ENGINE BUILD-UPS				$20,000	$200,000	$200,000	$420,000
TOTAL FABRICATION COSTS	$60,000	$240,000	$200,000	$70,000	$200,000	$200,000	$970,000
TOTAL FAB AND LABOR COSTS	$525,000	$914,000	$794,000	$588,000	$871,000	$730,000	$4,442,000

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NOTES:

1. ALL HARDWARE WOULD BE DEVELOPMENTAL AND OF BREAD-BOARD CONFIGURATION.
2. ALL TESTING WOULD BE PERFORMED IN GOVERNMENT FACILITIES WITH CONTRACTOR EMPLOYEES ASSISTING GOVERNMENT EMPLOYEES.

3.0 CONCLUSIONS AND RECOMMENDATIONS

Two undeveloped LOX/LH_2 rocket engine cycles were assessed, the split expander cycle and the full flow staged-combustion cycle.

The full flow staged-combustion cycle is so-termed because essentially the full flow of both propellants are used as turbine drive fluids. Two immediately obvious advantages are that turbine inlet temperatures can be low, and turbine drive fluids are compatible with the liquids being pumped so the need for positive dynamic shaft seals is eliminated. This configuration is also ammenable to high/variable mixture ratio operations. This engine cycle opens the way to many other benefits, for a LOX/LH_2 system, including: (1) A full flow topping cycle LOX/LH_2 engine which offers high Isp, large design margins, and low-cost components; (2) propellant turbopumps whose drive turbines run at room temperatures, and which have simple configurations; (3) preburners utilizing the concept of submerged combustion to ensure cold chamber wall temperatures and yet achieve complete combustion of the minor propellant; (4) Gas-gas main injectors, which also serve as the combustion chamber walls, inflow of the propellants serve as transpiration coolants, L^* is minimized, and efficient combustion is expected to be achieved by initiating combustion from the gaseous phase; and (5) A pre-mix main injector to achieve precise control of propellant mixing before they are admitted to the combustion zone. The split expander cycle is currently the subject of a demonstration program, and also exhibits many system advantages.

A technology development program has been formulated for the FFSC cycle to demonstrate the required technologies to develop a full scale flight engine in the 500K lbf thrust class in approximately eight years. Because of the cycle's insensitivity to component efficiencies, the engine can be scaled down to thrust sizes suited to STV uses and up to sizes appropriate for very heavy lift booster applications (greater than 1 million pounds of thrust). Development of enabling technologies potentially represents a high return on investment because of the potential for high payoff at a very low development risk.

4.0 REFERENCES

Many references for this effort are found within Figure 2.1-1, which served as the data base for the cycle identification and screening subtask. In addition to the references listed in the data base, a listing of the documents reviewed in the process of performing this effort is given below.

1. Space Transportation Engine Program, Engine Requirements Review, Pratt and Whitney, 18 October 1989, NAS8-38170.

2. STEP Engine Requirements Review, Rockwell International/Rocketdyne Division, 18 October 1989.

3. STEP Phase B Engine Requirements Quarterly Review, Gencorp/Aerojet Tech Systems, 19 October 1989.